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USATECOM PROJECT NO. 4-6-0500-01
USAASTA PROJECT NO. 66-06

ENGINEERING FLIGHT TEST
AH-1G HELICOPTER (HUEYCOBRA)

PHASE D

PART 2
PERFORMANCE

ADDENDUM

FINAL REPORT

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MARCH 1971

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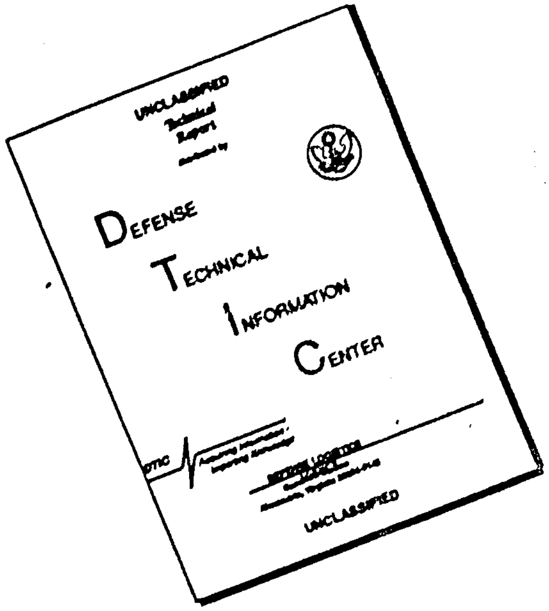

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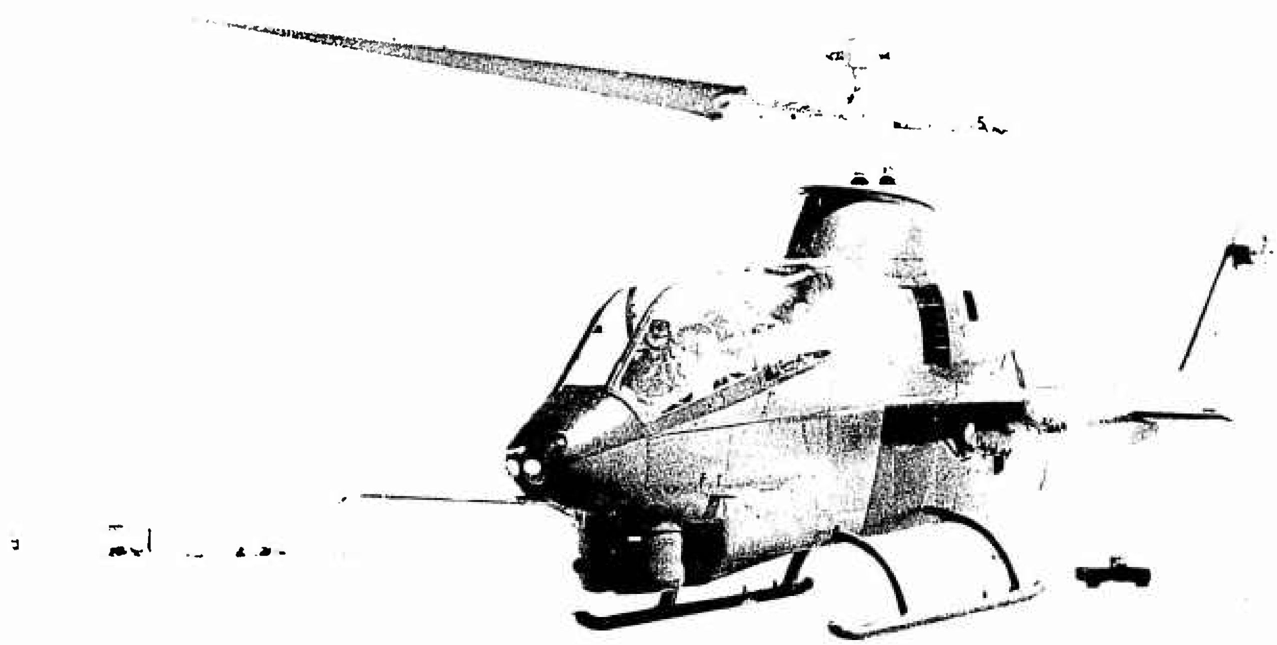

ABSTRACT

The Phase D Airworthiness and Qualification tests of the AH-1G helicopter were conducted in California at Edwards Air Force Base and auxiliary test sites during the period 13 June 1968 through 29 July 1969. This addendum to the performance report presents the results of turning performance, in-ground-effect (IGE) level acceleration and deceleration performance and dive recovery tests. These three tests were conducted to validate portions of the AH-1G operator's manual (TM 55-1520-221-10) and enhance the knowledge of interested government agencies as to the limitations and capabilities of the AH-1G helicopter. There were no additional deficiencies or shortcomings revealed by the results of these tests that had not been previously mentioned in Part 1, Part 2 and Part 3 of this report. Three major limitations were encountered during testing that restricted the pilot from achieving maximum performance: 1) level acceleration and deceleration performance IGE is limited by extreme pitch attitudes; 2) level deceleration performance is limited by the pilot's ability to maintain rotor speed below the maximum limit (339 rpm); 3) the cyclic control feedback limits aircraft turning and dive recovery performance at heavy gross weight and/or when high load factors are encountered.

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INTRODUCTION

BACKGROUND

1. In October 1965, the Department of the Army directed the US Army Materiel Command (USAMC) to conduct an expedited comparative evaluation of a selected group of three helicopters to fulfill the immediate requirement for an armed helicopter. A flight test program was conducted on the three aircraft by the US Army Aviation Systems Test Activity (USAASTA) at Edwards Air Force Base, California, from 13 November to 1 December 1965. The AH-1G HueyCobra was the aircraft selected from the evaluation to meet this requirement.

2. On 17 August 1966, USAASTA was directed by the US Army Test and Evaluation Command (USATECOM) to perform Phase B and Phase D testing of the AH-1G helicopter (ref 1, app I). A test plan for the Phase B engineering test was submitted by USAASTA in April 1967 and approved by the US Army Aviation Systems Command (USAAVSCOM). Phase B tests were conducted at different test sites and geographical locations from 3 April 1967 to 3 May 1968 on several test aircraft. The results of these tests are contained in references 2 through 8. The test plan for the Phase D program (ref 9) was initially submitted in August 1967 and was approved by USAAVSCOM on 24 October 1968. The Phase D test plan was amended on 5 November 1968 to include an additional test requested by USAAVSCOM (ref 10). Two aircraft were used for the Phase D test program to reduce the calendar testing time. One of the test aircraft was a prototype (aircraft S/N 66-15247), the other was a production model (aircraft S/N 67-15695). This addendum to the performance report contains test results for turning performance, level-flight acceleration and deceleration performance, and dive performance testing. The results of other performance tests are presented in the AH-1G Phase D, Part 2 report. The Phase D handling qualities and vibration characteristics are presented in the Phase D, Part 1 and Part 3 reports, respectively. No wing store jettison or armament subsystem firing tests were conducted during the Phase D program since adequate testing had been accomplished in these areas during the AH-1G Phase B program.

TEST OBJECTIVES

3. The objectives of the AH-1G Phase D test program were as follows:

a. To provide information for technical manuals and other service publications.

b. To determine compliance with applicable military specifications.

c. To determine compliance with contract guarantees.

d. To evaluate operational suitability for the armed helicopter mission.

DESCRIPTION

4. The AH-1G helicopter, manufactured by Bell Helicopter Company (BHC), was designed specifically to meet the US Army requirements for an armed helicopter. Tandem seating is provided for a two-man crew. The main rotor system is a two-bladed, semirigid, "door hinge" type with the stabilizer bar removed. A conventional antitorque rotor is located near the top of the vertical stabilizer. The AH-1G is equipped with a three-axis stability and control augmentation system (SCAS) to improve the aircraft's handling qualities. The helicopter is powered by a Lycoming T53-L-13 turboshaft engine rated at 1400 shaft horsepower (shp) at sea level (SL) under standard-day, uninstalled conditions. The engine is derated to 1100 shp due to the maximum torque limit of the helicopter's main transmission. Distinctive features of the AH-1G are the narrow fuselage (36 inches), the stub mid-wing with four external store stations and the integral chin turret. The flight control system is of the mechanical, hydraulically boosted, irreversible type with conventional helicopter controls in the aft cockpit (pilot station). The controls in the forward cockpit (copilot/gunner station) consist of conventional antitorque pedals and sidearm collective and cyclic controls. An electrically operated force trim system is connected to the cyclic and directional controls to induce artificial feel and to provide positive control centering. The elevator is synchronized with the longitudinal cyclic stick. The armament configurations are changed by varying the wing stores and flexible chin turret configurations. The pilot fires the wing stores and can fire the chin turret only in the stowed position. The copilot/gunner operates the flexible chin turret and can also fire the wing stores in an emergency. The wing stores can be jettisoned by either the pilot or copilot/gunner in case of emergency. The design gross weight (grwt) for the AH-1G is 6600 pounds, and the maximum grwt is 9500 pounds. More detailed aircraft information and operating limits of the AH-1G are presented in appendix II.

SCOPE OF TEST

5. During the AH-1G Phase D test program, 256 flights were conducted for a total of 368.8 hours of which 227.9 hours were productive test hours. Testing was conducted to determine the aircraft performance, handling qualities and vibration characteristics. Testing was conducted in California from 13 June 1968 through 29 July 1969 at Shafter Airport (420-foot elevation), Edwards AFB (2300-foot elevation) and at high-altitude test sites near Bishop (4120-, 7010- and 9500-foot elevations).

6. This addendum to the performance report contains the results of the turning performance, level-flight acceleration and deceleration performance and altitude loss during recovery from a dive. A total of 143.4 hours and 173 flights were required for all Phase D performance tests. The performance tests dicussed in this report required 10 flights for a total of 9.4 hours. All performance testing was accomplished on aircraft S/N 66-15247. The configurations tested during this portion of the program are listed in table 1.

Table 1. Aircraft Armament Configurations.[1]

Configuration	Armament Subsystem
Clean	TAT-102A or XM28 turret, no external wing stores
Heavy hog	TAT-102A or XM28 turret, two XM159 each wing

[1]The test aircraft was equipped with the TAT-102A chin turret: one 7.62 minigun (XM134).

7. The test program was conducted within the limitations established by the AH-1G safety-of-flight releases issued by USAAVSCOM, (refs 11 and 12, app I).

8. The empty weight of the test aircraft (S/N 66-15247) in a clean configuration with test instrumentation installed was 5790 pounds with a center of gravity (cg) at fuselage station (FS) 205.97.

9. The AH-1G was evaluated as an armed tactical helicopter, capable of day or night operation from prepared or unprepared areas. These three performance tests were conducted to validate portions of the AH-1G operator's manual (ref 13, app I) and enhance the knowledge

of interested government agencies as to the limitations and capabilities of the AH-1G helicopter. Specific test conditions for each test are presented in the Results and Discussion section of this addendum.

METHODS OF TEST

10. Test methods and data reduction procedures used in these tests were developed during the progress of the program, since established engineering flight test techniques were not available. The test methods and data reduction procedures are presented in appendix III. All flights were conducted in nonturbulent atmospheric conditions so the data would not be influenced by uncontrolled disturbances.

11. The flight test data were recorded from test instrumentation in the pilot panel, copilot/gunner panel, photopanel and 24-channel oscillograph. A detailed listing of the test instrumentation is included in appendix IV.

CHRONOLOGY

12. The chronology of the AH-1G Phase D, Part 2 test program is as listed:

Phase B flight test completed	3 May	1968
Phase D flight test commenced	13 June	1968
Phase D flight test completed	29 July	1969
Advance copy of report submitted	December	1970

RESULTS AND DISCUSSION

GENERAL

13. This addendum to the performance report presents the results of three engineering Phase D performance flight tests conducted on the AH-1G helicopter. These three tests are turning performance, in-ground-effect (IGE) level acceleration and deceleration performance and altitude loss during recovery from a dive. The tests were conducted to validate portions of the AH-1G operator's manual (ref 13, app I) and enhance the knowledge of interested government agencies as to the limitations and capabilities of the AH-1G helicopter. There were no additional deficiencies or shortcomings revealed as a result of these tests that were not reported in reference 14. The capability to perform some of these maneuvers was dictated by aircraft and piloting limitations. Three major limitations were encountered during testing that restricted AH-1G maneuvering performance: 1) level acceleration and deceleration performance IGE is limited by extreme pitch attitudes; 2) level deceleration performance is limited by the pilot's ability to maintain rotor speed below the maximum limit (339 rpm); 3) the cyclic control feedback limits aircraft turning and dive recovery performance at heavy gross weight and/or when high load factors are encountered.

AIRCRAFT CONTROL SYSTEM RIGGING

14. Prior to testing, the aircraft flight and engine controls were checked for correct rigging. Subsequent aircraft and engine control rigging changes were coordinated with contractor technical representatives.

ACCELERATION AND DECELERATION PERFORMANCE

15. The objective of these tests was to evaluate the level acceleration and deceleration capabilities of the AH-1G as a function of gross weight and wing store armament configurations. These tests were conducted at zero sideslip with the aircraft IGE at a skid height ranging from 5 to 10 feet. No attempt was made to conduct these tests out of ground effect (OGE) and/or at maximum sideslip (side flares). Handling qualities and vibration characteristics were qualitatively evaluated during each acceleration and deceleration test. Figures 1, 2 and 4, appendix V, show the rate of change

of aircraft energy during accelerations and decelerations, and figures 3, 5 and 6 present distances required to accelerate and decelerate. Time histories of several accelerations and decelerations are presented in figures 7 through 10. The conditions tested are presented in table 2.

Table 2. Level Acceleration and Deceleration Performance Test Conditions.[1]

Configuration	Average Gross Weight (lb)	Average Center of Gravity (in.)
Clean	8400, 7300	196.0 (mid)
Heavy hog	9300, 8400	195.5 (mid)

[1]Tests were conducted at a 500-foot density altitude (H_D) and a rotor speed of 324 rpm.

16. An energy method was used to analyze the level acceleration performance data. Since kinetic energy (E) equals 1/2 m V_t^2, where m is the mass of the aircraft and V_t is true airspeed, the rate of change in aircraft kinetic energy with respect to time can be expressed as:

$$\frac{dE}{dt} = 1/2 \; m \frac{d\left(V_t^{\,2}\right)}{dt}$$

Using this method of analysis, gross weight variations between different test conditions are automatically normalized for each armament configuration. This method of anlysis yields test results in terms of foot-pounds/second (ft-lb/sec) which can be converted to horsepower (hp). The airspeed at which the maximum rate of change in kinetic energy occurred was approximately 10 knots higher than airspeed for OGE level-flight minimum power required for the same test conditions. This variation in airspeed was attributed to ground effect and non-uniform application of engine power (para 17). The energy rate (dE/dt) was highest in the clean configuration (minimum equivalent flat plate area). Engine power output, as expected, had a significant effect on dE/dt and acceleration capability for the test technique employed. Figures 1 and 2, appendix V, graphically illustrate the decrease in aircraft energy rate of change and resulting acceleration capability when engine output power is decreased. One acceleration test run at 1124 shp (fig. 2) produced approximately 21,000 ft-lb/sec less rate of change in kinetic energy (equivalent to 38 shp) than the three other runs performed at an average engine power of 1192 shp. The fact that the engine power difference of 68 shp was not entirely converted to energy may be attributed to variations in rotor efficiency and/or lag and inaccuracy of the instrumentation and/or to limitations of the data reduction and analysis methods. The elapsed time required to accelerate to a given airspeed increased with increasing gross weight and/or larger values of equivalent flat plate area.

17. The piloting technique used to obtain the level acceleration is presented in appendix III. This technique required precise aircraft control inputs to maintain a constant altitude during each acceleration. Engine power was increased gradually as forward cyclic was applied. Engine power was stabilized 5 to 8 seconds after the start of the acceleration maneuver. At this time, a true airspeed of 60 to 70 knots was realized. The rate of power application was different for each acceleration, causing a different power/airspeed relationship. This variation resulted in data inconsistency between the entry airspeed (approximately 20 KTAS) and 75 KTAS since acceleration is directly related to the excess engine power output. The collective control inputs and engine power characteristics can be seen in figures 7 and 8, appendix V.

18. The handling qualities and vibration characteristics were acceptable during the level accelerations at all gross weights. However, when initiating a level acceleration at light gross weight (7300 lb), the coordination of pitch attitude change (18 to 25 degrees) and the application of engine power to maintain a constant skid height is a difficult piloting task. The severity of this pitch attitude and engine power coordination problem is illustrated in figure 7, appendix V.

19. The rate of change in aircraft energy during deceleration performance tests varied as the gross weight (disc loading) changed. This variation in deceleration performance was attributed to two related limitations: 1) the maximum rotor speed limit (339 rpm), and 2) the nose-up pitch attitude.

20. At the heavier gross weight (9300 lb), the rotor speed approached the maximum limit at a much faster rate. To prevent the rotor from overspeeding, the pilot had to constantly adjust collective control during the deceleration maneuver (fig. 10, app V). This continual adjustment of collective control caused the rotor thrust to vary and introduced a variation in the resulting deceleration.

21. The maximum rotor speed limit was not approached during deceleration performance at the lighter gross weight (7500 lb), and the rotor speed control was not as critical with respect to collective control inputs. The pitch attitude was the performance limiting parameter at the light gross weight. The pilot adjusted longitudinal cyclic to maintain the desired pitch attitude (fig. 9, app V). The pitch attitude range of 13 to 17 degrees was considered to be the maximum tolerable from an aircraft safety consideration. This attitude profile placed the tail boom skid within 2 to 4 feet of the ground for a skid height of 5 feet (closest point to the ground). Forward visibility was restricted by the airframe and was disconcerting to the pilot. Performing the maneuvers at higher skid heights would have allowed larger pitch attitude angles to be realized without danger of the tail boom skid contacting the ground. Higher pitch attitudes would also result in a higher rotor speed and higher deceleration performance.

22. The results of the deceleration performance tests were most consistent at a gross weight of 8400 pounds. The combination of a 13- to 17-degree nose-up pitch attitude and full-down collective resulted in a rotor speed equal to the maximum rotor limit. The control inputs (longitudinal and collective) required to maintain attitude and rotor speed throughout the deceleration maneuver were substantially less than at either heavier (9300 lb) or lighter (7500 lb) gross weights.

23. The handling qualities and vibration characteristics were acceptable during the level deceleration tests, except for pilot workload required to monitor critical parameters (rotor speed and pitch attitude). The monitoring task increased pilot workload but did not reach a point of degrading the handling qualities. If this task were performed in a confined area, it is possible that the accompanying additional human stress factor would increase the pilot workload to an unacceptable level.

24. Tail rotor power requirements to maintain zero sideslip during the terminal phase of the maneuver varied between 90 and 120 hp, depending on gross weight. The largest tail rotor horsepower value was encountered at a gross weight of 9300 pounds, the heaviest value tested.

TURNING PERFORMANCE

Teardrop Turns

25. The objectives of these tests were to determine time required to return to target when performing a coordinated "teardrop" turn maneuver without losing altitude and to reveal any handling qualities

or vibration limitations. This coordinated maneuver consisted of passing over a preselected point (target) on the ground, making a steep turn and returning over that spot in as short a time as possible. The maneuver was initiated by applying lateral cyclic control immediately after passing over the target on the ground. Maximum continuous engine power (50 psi) or maximum power available was applied as the desired roll attitude was approached. As the target returned to view, bank angle was reduced to a wings-level attitude, and the aircraft was accelerated back across the target. These maneuvers were performed to the left and right at each entry airspeed. The data from these tests are presented in figures 11 through 24, appendix V. The conditions tested are presented in table 3.

Table 3. Teardrop Turning Performance Test Conditions.[1]

Configuration	Average Gross Weight (lb)	Average Density Altitude (ft)	Average Center of Gravity (in.)	Trim Calibrated Airspeed (kt)
Clean	7700	3100	196.0 (mid)	$0.75V_H$, $0.9V_H$, V_H
Clean	8600	2400	193.5 (mid)	$0.85V_H$, V_H
Heavy hog	8650	2400	195.9 (mid)	$0.85V_H$, V_H
Heavy hog	9400	3000	196.3 (mid)	$0.85V_H$, V_H

[1]Tests were initiated from a trim rotor speed of 324 rpm.

26. When initiating the turn at 0.85 V_H, the quickest time (15 seconds) to accomplish the maneuver was at a light gross weight (7700 lb) in the clean configuration, and the longest time was at a heavy gross weight (9400 lb) in the heavy hog configuration. The high drag configuration reduced the acceleration and generally decreased the turning performance independently of gross weight for an entry airspeed range of 100 to 130 KCAS. The turning time was not significantly affected by the direction of the turn (right or left) at light gross weight (7700 lb) or medium gross weight (8600 lb). However, at heavy gross weight (9400 lb), a time of approximately 1 to 2 seconds longer was required to complete a left turn than a right turn, depending on entry airspeed.

27. The maximum roll rate and roll attitude encountered during these tests were generally found to be higher at the lighter gross weight (7700 lb). Bank angles in excess of 70 degrees and roll rates of 40 to 50 degrees per second (deg/sec) were common. These steep bank attitudes and high roll rates were easily controlled, and no special piloting technique was required. During the final portion of the turn, there was a tendency to develop a sudden rate of descent as the bank angle and turn rate were held constant and the load factor decreased with decreasing airspeed. At some reduced airspeed (40 to 50 KTAS) and at load factors less than 1.5 g's in the 60- to 70-degree banked turn, there was insufficient vertical thrust to maintain altitude, and a significant sink rate resulted. The aircraft feel was similar to that of an accelerated, no-buffet stall in a fixed-wing aircraft. Recovery was accomplished in approximately 100 feet by leveling the wings and increasing airspeed. The primary technique for avoiding this condition is to maintain indicated airspeed above 60 knots in a steeply banked turn. This condition was observed for both left and right turning maneuvers.

28. The maximum normal acceleration (load factor) ranged approximately from 2.0 to 2.6 g's. It was possible to achieve greater load factors at the lighter gross weight (7700 lb) than at the heavier gross weight (9400 lb). The wing armament stores configuration had no significant effect on the load factors obtained. A very reliable correlation was established between cyclic control force feedback and the numerical product of normal acceleration and gross weight of the aircraft. The pilot experienced heavy cyclic feedback when the product of these two parameters exceeded 18,500 to 19,000 pounds, while light cyclic force feedback was reported when this product was 16,500 to 17,000 pounds.

29. The handling qualities were acceptable during level "teardrop" turning maneuvers for all gross weights and wing store configurations tested. The onset of heavy cyclic feedback tended to cause the pilot to limit turning performance but did not jeopardize aircraft control. The vibration levels increased with increasing load factor during the turn but did not reach intolerable magnitudes.

30. Other return-to-target techniques were not investigated during the Phase D program. Results of "uncoordinated" level teardrop turns, climbing decelerating pedal turns, and descending teardrop turns are presented in references 15 and 16, appendix I.

<u>180-Degree Level Turns</u>

31. The primary objective of the 180-degree level turns was to determine the time required to turn at various entry airspeeds and then accelerate in the opposite direction to an airspeed of

$0.95V_H$. The secondary objective was to determine penetration distance and horizontal distance traversed during a turn as a function of heading change and entry airspeed for various gross weights (fig. A). The handling qualities and vibration characteristics were also evaluated to determine limitations on the aircraft's or pilot's ability to perform the maneuver. The results of these tests are presented graphically in figures 15 through 23, appendix V. Representative time histories are presented in figures 24 and 25. The test conditions are presented in table 4. The technique used was to cross a ground reference point at the selected stabilized entry airspeed and then rapidly roll the aircraft into a banked turn. Engine power was then increased by applying collective control following the initial lateral control input. Load factor increased as the aircraft progressed through the turn. As the reciprocal heading was approached, the wings were leveled and engine power was adjusted to the maximum permitted (50 psi) or the maximum engine topping power available.

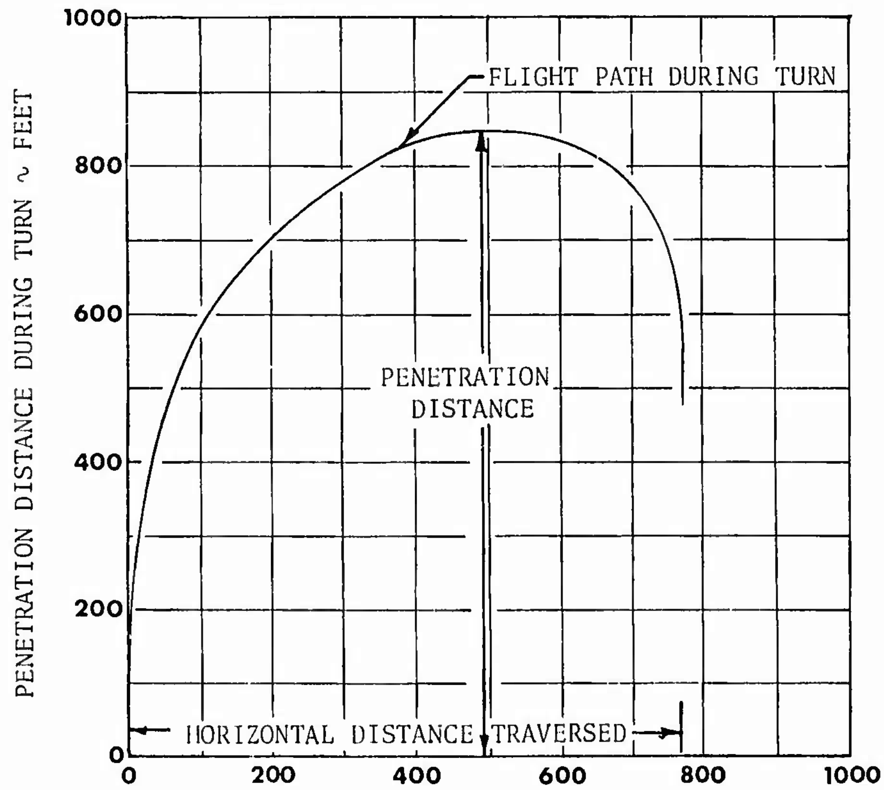

Figure A. 180 Degree Turning Performance.

Table 4. 180-Degree Level Turning Performance Test Conditions.[1]

Configuration	Average Gross Weight (lb)	Average Density Altitude (ft)	Average Longitudinal Center of Gravity (in.)	True Entry Airspeed Range (kt)
Clean	7250	3100	195.6 (mid)	90 to V_H
Clean	8250	2400	193.1 (mid)	90 to V_H
Heavy hog	8150	2400	195.6 (mid)	90 to V_H
Heavy hog	9100	3000	196.2 (mid)	90 to V_H

[1]Tests were initiated from a trim rotor speed of 324 rpm.

32. The time required to complete the maneuver increased significantly with gross weight and was only slightly affected by the wing store armament configuration. This time was 40 seconds at the lighter gross weight (7250 lb) and 49.5 seconds at the heavier gross weight (9100 lb) for an entry airspeed of 108 KTAS. The engine power available during each acceleration was approximately the same (within 15 shp) and should not have affected the difference in the acceleration times. There was a small variation in the time requirement with entry airspeed. No difference was noted in the time required to perform the maneuver either to the right or to the left.

33. The time required to accomplish the 180-degree heading change portion of the maneuver varied from 9.5 to 12 seconds, depending on the gross weight. This time required generally increased with increasing entry airspeed, while the wing store armament configuration had an insignificant effect.

34. The penetration distance during a 180-degree turn varied slightly for each combination of gross weight and wing store configuration. However, the trade-off between increased equivalent flat plate area or aerodynamic drag (caused by wing store configuration) and load factor achievement capability (dictated by gross weight) balanced and resulted in about the same penetration distances for a given entry airspeed. There was little difference in penetration distances between a right and a left turn. The data presented in figures 15 through 18, appendix V, are for a zero wind condition. The flight trajectory and penetration distances shown in these plots can be strongly influenced by wind. Paragraph 8-24 and figures 8-1 and 8-2, presenting radius-of-turn information in the AH-1G operator's manual (ref 13, app I), should be revised to reflect the test results and flight trajectory patterns presented in this report.

35. The maximum roll rates and roll attitudes realized during these tests were similar to those encountered during the teardrop turn maneuvers (para 27). For a given gross weight, the maximum load factor encountered during the turn increased with increasing entry airspeed. The peak normal accelerations encountered were generally less than the values measured during the teardrop turns. These lower load factors were probably the reason why cyclic control force feedback was limited to the "light" classification. The handling qualities were acceptable during the performance of this maneuver. Careful monitoring of the engine torque was required to avoid exceeding the torque limits, particularly during the roll-out from right turns.

36. A roll rate oscillation was noted during the turning performance tests (teardrops and 180-degree turns). Time histories of these oscillations are presented in figures 24 and 25, appendix V. The frequency of this roll rate oscillation was 1.35 Hertz (Hz) and reached a magnitude of ±12 deg/sec. This frequency is less than the 2.0 to 2.3 Hz of the engine speed governor oscillation reported in references 16 and 17, appendix I. This roll rate was sensed by both the test instrumentation and the stability and control augmentation system (SCAS). All tests were conducted with roll SCAS operating, and no attempt was made to determine if the SCAS inputs were a driving or damping function. The oscillation was sensed by the crew members but was not objectionable since there was little or no attitude change.

DIVE RECOVERY PERFORMANCE

37. The objective of this test was to determine the altitude loss during recovery from a dive as a function of flight path angle, normal acceleration and rate of descent. The handling qualities and vibration characteristics were also evaluated to reveal the limiting factors that might curtail the recovery capability. The results of these tests are presented in figures 26 through 32, appendix V. Time histories of these maneuvers are presented in figures 33 through 36. The conditions tested are shown in table 5.

38. These tests consisted of a series of dives at different rates of descent to determine altitude loss during a pull-out as a function of load factor. Each dive was initiated from a trimmed level-flight condition. With power control fixed, the aircraft was placed in a nose-down attitude and allowed to accelerate to the desired airspeed. The flight path angle was then adjusted to maintain this airspeed until the recovery altitude was reached. The pull-up was accomplished by applying aft cyclic control. The collective control and throttle control were not changed from the trim level-flight condition until after the dive and pull-out were completed.

Table 5. Dive Recovery Performance Test Conditions.[1]

Configuration	Average Gross Weight (lb)	Average Density Altitude (ft)	Average Longitudinal Center of Gravity (in.)
Heavy hog	8300	4300	196.3 (mid)
Heavy hog	8200	4700	195.7 (mid)
Heavy hog	9200	4700	195.7 (mid)
Clean	8200	5700	194.6 (mid)

[1]Tests were initiated from a trim rotor speed of 324 rpm.

39. The altitude loss during recovery varied with rate of descent, angle of descent and normal acceleration. The effect of flight path angle on altitude loss could not be determined precisely since the flight path angle for the conditions tested was less than 15 degrees (referenced to the horizontal). Calculated altitude loss for larger descent angles is shown in figure 32, appendix V. Gross weight variations had little affect on the altitude loss during this maneuver.

40. The maximum normal acceleration achieved during these tests varied from 2.01 to 2.16 g's depending on the gross weight of the aircraft. The data presented in figure 26, appendix V, indicate 175 feet in altitude would be lost during recovery with the helicopter descending at a rate of 5000 feet per minute if 2.0 g's were developed during pull-out.

41. There was an increase in engine power output as airspeed was increased during a dive at a constant collective setting as shown in table 6 and also in figures 33 through 36, appendix V. This increase in engine power output was not a problem during this test program since all diving maneuvers were entered at altitudes well above that where the maximum power limit (1100 shp) could be developed. However, this engine power increase could be a problem if the maneuver were entered at or near the torque limit of the main transmission. Continual monitoring of the engine torquemeter would be required to avoid exceeding the main transmission torque limit. This characteristic was previously qualitatively analyzed in reference 2, appendix I.

Table 6. Engine Shaft Horsepower Increase in Diving Flight.

Gross Weight (lb)	Collective Control (in. from full down)	Calibrated Entry Airspeed (kt)	Entry Engine Horsepower (shp)	Entry Density Altitude (ft)	Calibrated Exit[1] Airspeed (kt)	Exit[1] Engine Horsepower (shp)	Exit[1] Density Altitude (ft)
7710	2.65	65	490	9340	178	625	4695
7995	3.51	110	985	9255	180	1065	4810
8415	3.85	126	923	7660	180	1023	4820
9255	3.85	118	895	7200	180	1075	4170

[1]Exit values were averaged for approximately 2 seconds prior to pull-up maneuver.

42. There was a directional control input required to maintain zero sideslip angle in diving flight. This control input increased with increasing airspeed during the dive. The directional control requirement tended to increase at light gross weight and was maximum at a gross weight of 7710 pounds. The pilot effort required to maintain zero sideslip and to adequately accomplish the test objectives was not excessive. However, when precise target tracking is performed, this characteristic will increase pilot workload.

43. Cyclic control feedback was encountered when performing a maximum performance dive recovery maneuver. The severity and time duration of the cyclic feedback was dependent on the numerical product of gross weight and cg normal acceleration (para 28). Cyclic feedback and its characteristics during this maneuver should be annotated in the AH-1G operator's manual (ref 13, app I).

44. Paragraph 8-26 and figure 8-3 of the operator's manual (ref 13, app I) should be revised to reflect the flight conditions necessary to achieve the diving performance presented in this report. These conditions are as listed:

a. Zero acceleration along the flight path (stabilized airspeed and rate of descent).

b. Flight path angle less than 15 degrees (referenced to the horizontal).

c. Wings level during the pull-out maneuver.

45. All dive recoveries were initiated from an unaccelerated wings-level attitude. After extensive consultation with combat experienced AH-1G pilots, it was determined that the accelerated flight condition is encountered much more often than unaccelerated flight during diving flight maneuvers. No attempt was made during this program to determine the effects of accelerated diving flight and/or changing roll attitude (during recovery) on attitude loss when performing this maneuver. Additional testing is required to determine the effects of these two parameters.

CONCLUSIONS

46. The level acceleration performance IGE is limited by the extreme aircraft nose-down pitch attitudes (18 to 25 degrees) required to achieve maximum acceleration performance (para 18).

47. The level deceleration performance IGE is limited by the pilot's ability to control nose-up pitch attitude and maintain rotor speed below the maximum rotor limit (339 rpm) (paras 19, 20 and 21).

48. The forward visibility of the pilot is reduced when performing a maximum deceleration (para 21).

49. An undesirable rate of descent can develop during steeply banked high-performance decelerating turns at low airspeeds (para 27).

50. Cyclic control feedback will be encountered during maneuvers at a high numerical product of gross weight and load factor (paras 28, 29, 35 and 43).

51. Information on altitude loss during diving flight, presented in this report, is valid only for certain limiting conditions (para 44).

52. Additional testing is required to determine the effects of accelerating diving flight and changing roll attitude (during pull-out) on alitutde loss (para 45).

RECOMMENDATIONS

53. The operator's manual should be revised to include the graphic results and the limitations presented in this report (paras 18, 19, 27, 34 and 44).

54. Additional testing should be performed to determine the effects of accelerated diving flight and changing roll attitude (during pull-out) on altitude loss (para 45).

APPENDIX I. REFERENCES

1. Letter, AMSTE-BG, USATECOM, 17 August 1966, subject: Test Directive, Engineering and Logistical Evaluation Test of the AH-1G Helicopter (HueyCobra) (U).

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APPENDIX II. BASIC AIRCRAFT INFORMATION AND OPERATING LIMITS

AIRFRAME

Rotor System

1. The 540 "door hinge" main rotor assembly is a two-bladed, semi-rigid, underslung feathering-axis type rotor. The assembly consists basically of two all-metal blades, blade grips, yoke extensions, yoke trunnion, and rotating controls. Control horns for cyclic and collective control input are mounted on the trailing edge of the blade grip. Trunnion bearings permit rotor flapping. The blade grip-to-yoke extension bearings permit cyclic and collective pitch action.

Tail Rotor

2. The tail rotor is a two-bladed, delta-hinge type employing pre-coning and underslinging. The blade and yoke assembly is mounted to the tail rotor shaft by means of a delta-hinge trunnion. Blade pitch angle is varied by movement of the tail rotor control pedals. Power to drive the tail rotor is supplied by a takeoff on the lower end of the main transmission.

Transmission System

3. The transmission is mounted forward of the engine and coupled to the engine by a short drive shaft. The transmission is basically a reduction gear box which transmits engine power at reduced rpm to the main and tail rotors by means of a two-stage planetary gear train. The transmission incorporates a free-wheeling clutch unit at the input drive. This provides a disconnect from the engine in case of a power failure to allow the aircraft to make an autorotational landing.

Synchronized Elevator

4. The synchronized elevator, which has an inverted airfoil section, is located near the aft end of the tail boom and is connected by control tubes and mechanical linkage to the fore and aft cyclic control system. Fore and aft movements of the cyclic control stick produce a change in the synchronized elevator attitude.

Control Systems

5. A dual hydraulic control system is provided for the cyclic and collective controls. The directional controls are powered by a

single servo cylinder which is operated by system number 1. The hydraulic system consists of two hydraulic pumps, two reservoirs, reliev valves, shut-off valves, pressure warning lights, lines, fittings, and manual dual-tandem servo actuators incorporating irreversible valves. Tandem power cylinders incorporating closed-center four-way manual servo valves and irreversible valves are provided in the lateral, fore and aft cyclic and collective control system. A single power cylinder incorporating a closed-center four-way manual servo valve is provided in the directional control system. The cylinders contain a straight-through mechanical linkage.

Force Trim

6. Magnetic brake and force gradient devices are incorporated in the cyclic control and directional pedal controls. These devices are installed in the flight control system between the cyclic stick and the hydraulic power cylinders and between the directional pedals and the hydraulic power cylinder. The force trim control can be turned off by depressing the left button on the top of the cyclic stick. The gradient is accomplished by springs and magnetic brake release assemblies which enable the pilot to trim the controls as desired.

Cyclic Control Stick

7. The pilot and gunner cyclic stick grips each have a force trim switch and a SCAS release switch. The pilot cyclic stick has a built-in operating friction. The cyclic control movements are transmitted directly to the swash plate. The fore and aft cyclic control linkage is routed from the cyclic stick through the SCAS actuator, to the dual boost hydraulic actuator, and then to the right horn of the fixed swash plate ring. The lateral cyclic is similarly routed to the left horn.

Collective Pitch Control

8. The collective pitch control is located to the left of the pilot and is used to control the vertical mode of flight. Operating friction can be induced into the control lever by hand-tightening the friction adjuster. The pilot and gunner collective pitch controls have a rotating grip-type throttle.

Tail Rotor Pitch Control Pedals

9. Tail rotor pitch control pedals alter the pitch of the tail rotor blades and thereby provide the means for directional control. The force trim system is connected to the directional controls and is operated by the force trim switch on the cyclic control grip.

Stability and Control Augmentation System

10. The SCAS is a three-axis, limited-authority, rate-referenced stability augmentation system. It includes an electrical input which augments the pilot mechanical control input. This system permits separate consideration of airframe displacementc caused by external disturbances from displacements caused by pilot input. The SCAS is integrated into the fore, aft, lateral and directional flight controls to improve the stability and handling qualities of the helicopter. The system consists of electro-hydraulic servo actuators, control motion transducers, a sensor/amplifier unit and a control panel. The servo actuator movements are not felt by the pilot. The actuators are limited to a 25-percent authority and will center and lock in case of an electrical and/or a hydraulic failure.

ENGINE

Engine Description

11. The T53-L-13 engine, rated at 1400 shp, is a successor to the T53-L-11 engine. The additional power has been achieved with no change in the basic T53-L-11 engine envelope mounting and connection points and with a 6-percent increase in basic engine weight.

12. The performance gain is accomplished thermodynamically by the mechanical integration of a modified axial compressor, a two-stage compressor turbine and a two-stage power turbine into the T53-L-11 engine configuration.

13. Replacement of the first two compressor stators and changing of the first two stages of compressor rotor blades and discs results in an approximate 20-percent increase in mass air flow through the engine. This is accomplished without the use of inlet guide vanes.

14. An inlet flow fence, located on the outer wall of the inlet housing in the area of the previously used inlet guide vanes, provides the desired inlet conditions for the transonic compression during acceleration at low speeds. At compressor speeds up to 70 percent, the fence is in the extended position. Above 70 percent, the flow fence is retracted into the outer wall of the inlet housing. Similar to a piston ring, the circumference of the flow fence is changed by the action of a piston actuator powered by compressor discharge pressure.

15. The specification for this engine allows the use of JP-4 or JP-5 fuel for satisfactory operation throughout the engine's operating envelope. During this program, JP-4 fuel was used.

Engine Power Control System

16. The fuel control for the T53-L-13 engine is a hydro-mechanical type of fuel control. It consists of the following main units:

a. Duel-element fuel pump.

b. Gas producer speed governor.

c. Power turbine speed topping governor.

d. Acceleration and deceleration control.

e. Fuel shut-off valve.

f. Transient air bleed control.

17. An air bleed control is incorporated within the fuel control to provide for opening and closing the compressor interstage air bleed in response to the following signals present in the power control:

a. Gas producer speed.

b. Compressor inlet air temperature.

c. Fuel flow.

18. The fuel control is designed to be operated either automatically or in an emergency mode. In the emergency position, fuel flow is terminated to the main metering valve and is routed to the manual (emergency) metering and dump valve assembly. While in the emergency mode, fuel flow to the engine is controlled by the position of the manual metering valve which is connected directly to the power control (twist grip). During the emergency operation, there is no automatic control of fuel flow during acceleration and deceleration; thus, engine acceleration and exhaust gas temperature (EGT) must be pilot monitored.

BASIC AIRCRAFT INFORMATION

Airframe Data

Overall length (rotor turning)	637.2 in.
Overall width (rotor trailing)	124.0 in.
Centerline of main rotor to centerline of tail rotor	320.7 in.
Centerline of main rotor to elevator hinge line	198.6 in.
Elevator area (total)	15.2 sq ft
Elevator area (both panels)	10.9 sq ft
Elevator airfoil section	Inverted Clark Y
Vertical stabilizer area	18.5 sq ft
Vertical stabilizer airfoil section	Special camber
Vertical stabilizer aerodynamic center	FS 499.0
Wing area:	
Total	27.8 sq ft
Outboard of butt line (BL) 18.0 (both sides)	18.5 sq ft
Wing span	10.33 ft
Wing airfoil section:	
Root	NACA 0030
Tip	NACA 0024
Wing angle of incidence	14 deg

Main Rotor Data

Number of blades	2
Diameter	44 ft
Disc area	1520.5 sq ft

Blade chord	27 in.
Rotor solidity	0.0651
Blade area (both blades)	99 sq ft
Blade airfoil	9.33 percent symm special section
Linear blade twist	-0.455 deg/ft
Hub precone angle	2.75 deg
Rotor inertia	2900 slug-ft^2

Antitorque Rotor Data

Number of blades	2
Diameter	8.5 ft
Disc area	56.74 sq ft
Blade chord	8.41 in.
Rotor solidity	0.105
Blade airfoil	NACA 0010 modified
Blade twist	Zero deg

Transmission Drive System Ratios

Engine to main rotor	20.383:1.0
Engine to antitorque rotor	3.990:1.0
Engine to antitorque drive system	1.535:1.0

Test Aircraft Control Displacements

Longitudinal cyclic control:

Full forward to full aft with SCAS nulled	9.07 in.

Lateral cyclic control:

Full left to full right with SCAS nulled	10.00 in.

Directional (pedal) control:

Full left to full right with SCAS nulled	7.07 in.

Collective control:

Full up to full down with SCAS nulled	9.30 in.

OPERATING LIMITATIONS

Limit Airspeed

Any configuration with XM159 rocket pods:

180 KCAS below a 3000-foot H_D; decrease 8 KCAS per 1000 feet above 3000 feet

For this test, the AH-1G with skid gear fairings removed:

Same as standard configurations (normal limit for operational use: 160 KCAS)

All other configurations:

190 KCAS below a 4000-foot H_D; decrease 8 KCAS per 1000 feet above 4000 feet

Gross-Weight/Center-of-Gravity Envelope

Forward cg limit:

Below 7000 pounds, FS 190.0; linear increase to FS 192.1 at 9500 pounds

Aft cg limit:

Below 8270 pounds, FS 201.0; linear decrease to FS 200 at 9500 pounds

Sideslip Limits

Five degrees at V_L with linear increase to 30 degrees at 50 KCAS

Rotor and Engine Speed Limits (Steady State)

Power on:	
Engine rpm	6400 to 6600
Rotor rpm	314 to 324
Power off:	
Rotor rpm	294 to 339
Rotor rpm transient lower limit	250
Power on during dives and maneuvers:	
Rotor rpm	314 to 324

Temperature and Pressure Limits

Engine oil temperature	93°C
Transmission oil temperature	110°C
Engine oil pressure	25 to 100 psi
Transmission oil pressure	30 to 70 psi
Fuel pressure	5 to 20 psi

T53-L-13 Engine Limits

Normal rated EGT (maximum continuous)	625°C
Military rated EGT (30-minute limit)	645°C
Starting and acceleration EGT (5-second limit)	675°C
Maximum EGT for starting and acceleration	760°C
Torque pressure limit	50 psi

APPENDIX III. TEST TECHNIQUES AND DATA REDUCTION PROCEDURES

INTRODUCTION

1. The test techniques and data reduction procedures used to obtain and standardize test results (where possible) were developed during the course of these tests since little or no flight testing had been performed in these areas with rotary-wing aircraft. In many cases, fixed-wing techniques were applied, and where repeatable results were obtained, these techniques of operation and data reduction were adopted. In areas where fixed-wing methods did not yield repeatable data, either modified fixed-wing or new techniques were developed.

INSTRUMENTATION

2. All instrumentation was calibrated prior to commencing the test program. A detailed tabulation of the instrumentation is given in appendix IV. All quantitative data obtained during this test program were derived from special sensitive instrumentation located on the aircraft or on the ground. The aircraft sources were the oscillograph, photopanel, pilot panel (hand recorded) and the copilot/gunner panel (hand recorded). The ground support sources were the ground station, Fairchild camera station and four Askania cinetheodolite cameras. The ground station and Fairchild camera station were manned and supported by USAASTA personnel. The Askania cinetheodolite cameras were operated by the US Air Force (USAF) Space Positioning Branch personnel from Edwards AFB. The Askania cinetheodolite camera film was read by the USAF Data Systems Branch personnel and reduced on an IBM 7094 digital computer.

WEIGHT AND BALANCE

3. The weight and balance of the test helicopter was carefully maintained during the test program. Variations in empty weight and cg due to changes of helicopter components and/or instrumentation were defined by periodic weighings.

4. The empty weight of the test aircraft without instrumentation installed could not be determined since the aircraft was partially instrumented when it was delivered to USAASTA at the beginning of the program. In addition, the aircraft was not a production model and was not representative of a standard AH-1G. The fuel load of the aircraft was defined by measuring the fuel specific gravity and temperature after fueling, and by using an external sight gage on

the calibrated fuel cell to determine fuel volume. Fuel used in flight was recorded by a calibrated fuel-used system and the results were cross-checked with the sight gage reading following each flight. Helicopter loading and cg were controlled by ballast at various locations in the aircraft.

LEVEL ACCELERATIONS AND DECELERATIONS

5. Level accelerations were initiated from a stabilized calibrated airspeed of approximately 20 KCAS. The altitude (5-foot skid height) was maintained with cyclic pitch, and the aircraft was allowed to accelerate as engine power was increased to 50 psi (indicated torque) and stabilized. When engine speed and rotor transients were stabilized, a final adjustment to realize maximum power was made. The tests were performed over a surveyed course and recorded with a Fairchild flight analyzer.

6. Level decelerations using zero sideslip were accomplished at entry true airspeeds varying between 30 and 110 knots. It was determined that the best technique was to initiate an aft cyclic flare and collective control reduction simultaneously at the start of the maneuver. An engine-rotor needle split was realized as the nose of the aircraft pitched up and the engine power output decreased. To prevent the rotor speed from exceeding the upper rotor limit and to maintain a constant altitude, the rate of aft cyclic control and down collective control application had to be well coordinated. At about 30 KTAS, the collective was increased to establish a hover at the end of the maneuver. The altitude throughout this maneuver was maintained at or near a 5-foot skid height.

7. The acceleration and deceleration performance was analyzed using a simplified energy method. This energy method was derived from Newton's second law. The Fairchild flight data were used to determine horizontal distance and vertical height as a function of time. Since the vertical height did not change significantly during the acceleration or deceleration tests, potential energy was excluded from the analysis. Faired values of horizontal distance versus related time were then analyzed by use of an IBM 1620 computer to derive the coefficients for a polynomial equation that was representative of the input values. A least squares method was employed as the curve fitting technique. The computer analysis yields distance (x) as a function of time (t) or:

$$\text{Distance:} \quad x = f(t) \tag{1}$$

$$\text{Velocity:} \quad \frac{dx}{dt} = \frac{d\ f(t)}{dt} \tag{2}$$

$$\text{Acceleration:} \quad \frac{d^2x}{dt^2} = \frac{d^2\ f(t)}{dt^2} \tag{3}$$

Applying Newton's second law to determine the kinetic energy (E) of a body yeilds:

$$E = 1/2\ m\ V_t^{\ 2} = 1/2\ m \left(\frac{dx}{dt}^{2}\right) \tag{4}$$

The rate of change of kinetic energy is thus equal to:

$$\frac{d(E)}{dt} = 1/2\ m\ \frac{d\left(\frac{dx}{dt}^{2}\right)}{dt} = m\ \frac{dx}{dt}\ \frac{d^2x}{dt^2} \tag{5}$$

where: $m = \frac{GRWT}{g}$ $\qquad g = 32.17\ ft/sec^2$

The highest order polynomial that was considered to be acceptable was the fourth order or in some cases a fifth order. Polynomials of higher order yielded unsatisfactory acceleration terms.

8. This simplified method did not include the variations of rotor speed during the acceleration and deceleration maneuvers or the kinetic energy of rotation about the mass center. The test data was not standardized for variations in density altitude and engine shaft horsepower output.

TEARDROP TURNS

9. This coordinated maneuver consisted of passing over a preselected point (target) on the ground, making a steep turn either left or right and returning over that spot in as short a time as possible, maintaining constant altitude. The maneuver was initiated by applying lateral cyclic control immediately after passing over the target on the ground. Maximum continuous engine power (50 psi) or engine topping power was applied as the desired roll attitude was approached. As the target came back into view, bank angle was reduced to a wings-level attitude, and the aircraft was accelerated back across the target. Small engine power adjustments were required during the accelerating return to the target to avoid

exceeding the torque limit. Time required to accomplish each maneuver was recorded both in the aircraft and by ground observers. Each entry airspeed was performed on reciprocal headings to average wind effects.

10. No attempt was made to standardize these data since the maneuver is transient in nature and the overall results are a function of pilot proficiency, individual aircraft performance, handling qualities, and vibration characteristics.

180-DEGREE TURNS

11. These tests consisted of performing a series of 180-degree turns at different entry airspeeds followed by an acceleration to $0.95V_H$, maintaining a constant altitude. The entry and engine power application procedure was similar to that of the teardrop turn. However, the total heading change during the maneuver was limited to 180 degrees. Engine power output was adjusted during the accelerating portion of each maneuver to avoid exceeding the engine torque limit.

12. The turning portion of this maneuver was transient, and the final results were predicated on the same intangible variables mentioned in paragraph 10. No attempt was made to standardize the test data for this reason.

13. The penetration distance as a function of entry airspeed, heading change and gross weight was calculated by using the following equations:

$$\Delta \text{ Heading Change} = \frac{57.3g \sqrt{n_z^2 - 1} \times \Delta T}{V_t \times 1.688} \quad (6)$$

$$\Delta \text{ Penetration Distance} = V_t \times \cos (\Sigma\Delta \text{ Heading Change}) \times \Delta T \quad (7)$$

14. The radius of turn as a function of airspeed was determined by the following equation:

$$R = \frac{V_t^2 \times (1.688)^2}{g\sqrt{n_z^2 - 1}} \quad (8)$$

DIVE RECOVERY PERFORMANCE

15. These tests consisted of performing a series of dives at different rates of descent to determine altitude loss during a pull-out from a dive as a function of load factor. The dive was initiated from a trimmed level-flight condition over a predesignated ground path so a continuous record of altitude could be recorded by four Askania cinetheodolite cameras during each test point. The collective control and throttle control were not changed from the trim level-flight condition until after the dive and pull-out were completed. The information gathered by the four Askania cinetheodolite cameras was used to obtain instantaneous altitude and rate of descent by a highly complex triangulation method. The use of the Askania cinetheodolite cameras allowed an accurate determination of these two parameters without error being introduced by pitot-static system lags.

16. The altitude loss during recovery from a dive was standardized by the following equation:

$$\Delta H = \frac{(R/D)^2}{(1 + \cos\gamma)\ g\ (n_z - 1)} + (R/D \times \Delta T) \qquad (9)$$

where: ΔT = The average time required to establish the desired pitch rate (0.81 sec)

R/D = Feet per second

$g = 32.17\ ft/sec^2$

17. This time increment (ΔT) was determined by curve fitting various values to the test data. A value of 0.81 second was found to yield the most repeatable results for the conditions tested.

APPENDIX IV. TEST INSTRUMENTATION

Flight test instrumentation was installed in the test helicopter prior to the start of this evaluation. This instrumentation provided data from four sources: pilot panel (photo 1), engineer panel (photo 2), photopanel (photo 3), and a 24-channel oscillograph (photo 4). All instrumentation was calibrated. The flight test instrumentation was installed and maintained by USAASTA. The following test parameters were presented.

PILOT PANEL

Standard system airspeed
Boom system airspeed
Boom system altitude
Rate of climb
Gas producer speed
Torque pressure (standard system)
Exhaust gas temperature
Longitudinal control position
Lateral control position
Pedal control position
CG (normal acceleration)
Angle of sideslip

ENGINEER PANEL

Boom system airspeed
Boom system altitude
Outside air temperature
Rotor speed
Gas producer speed
Fuel used (total)
Torque pressure (high)
Torque pressure (low)
Exhaust gas temperature
Oscillograph correlation counter
Photopanel correlation counter
Fuel temperature
Engine fuel flow

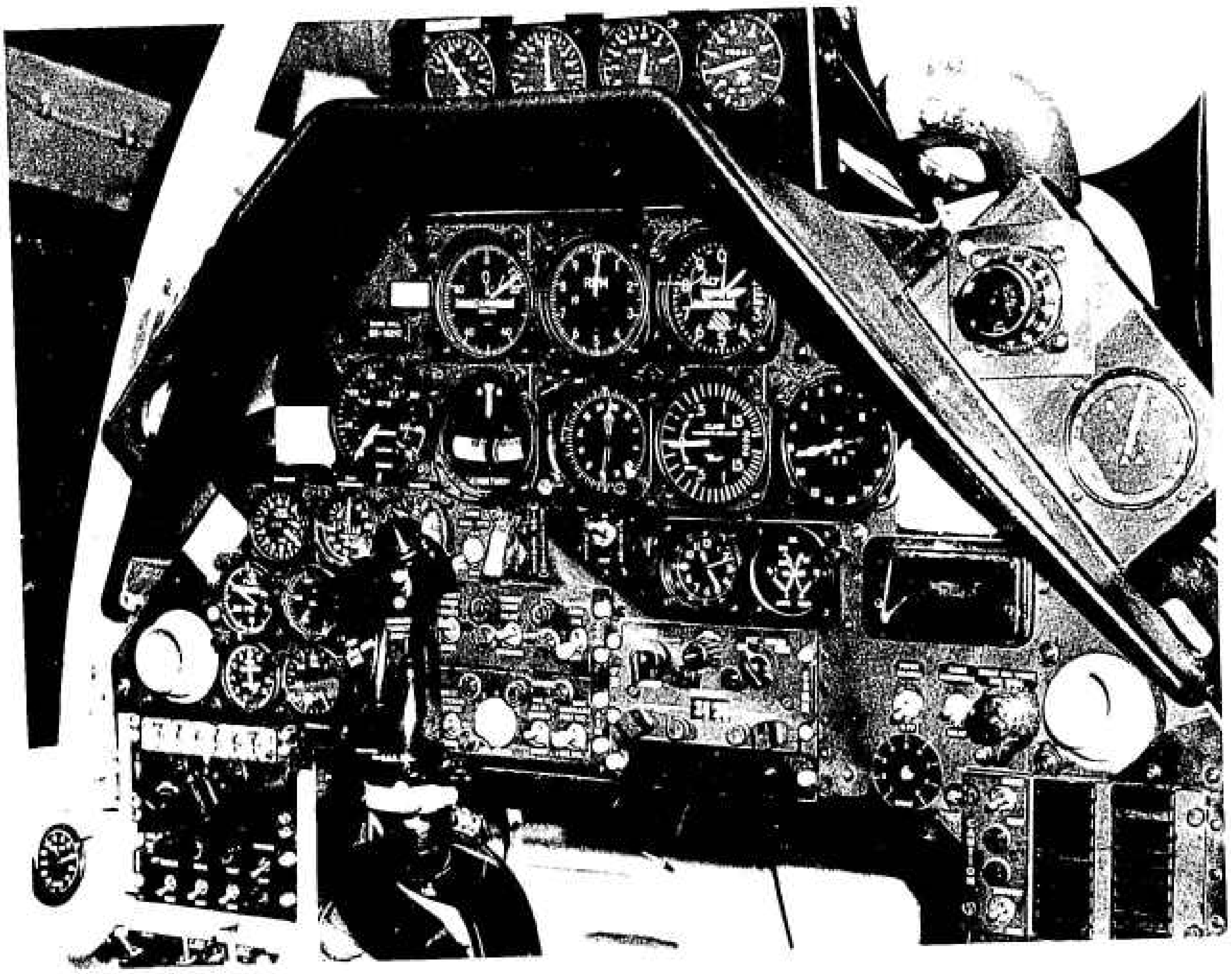


Photo 1. Pilot Panel.

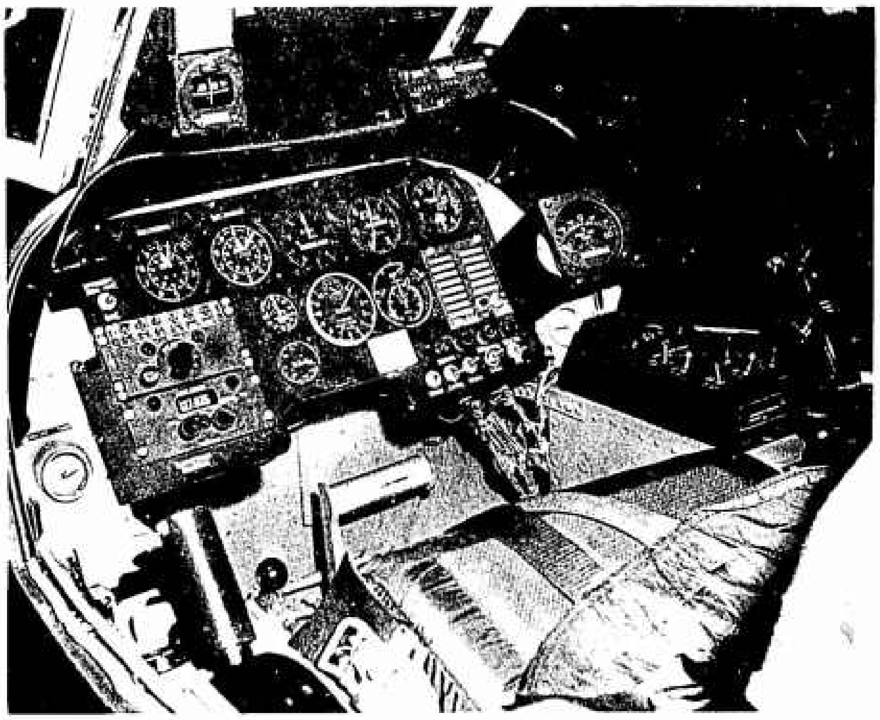


Photo 2. Copilot/Engineer Panel.

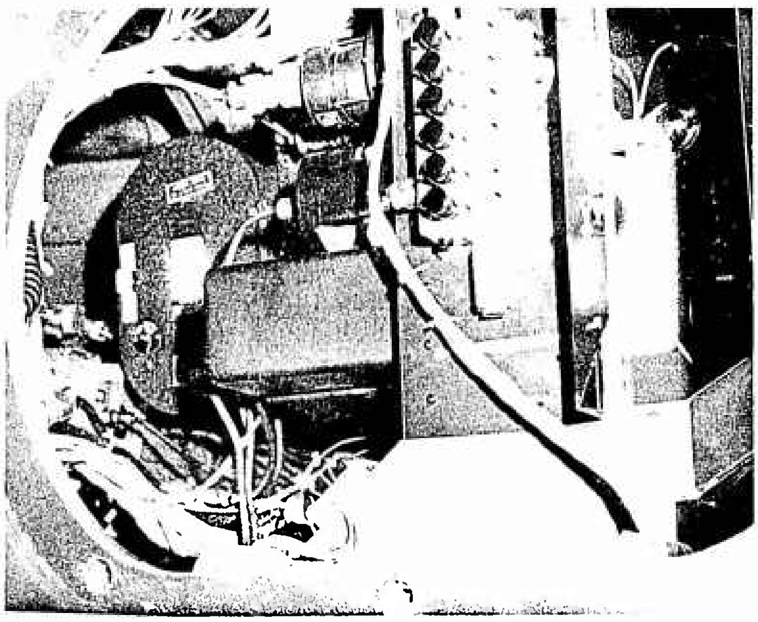


Photo 3. Photopanel.

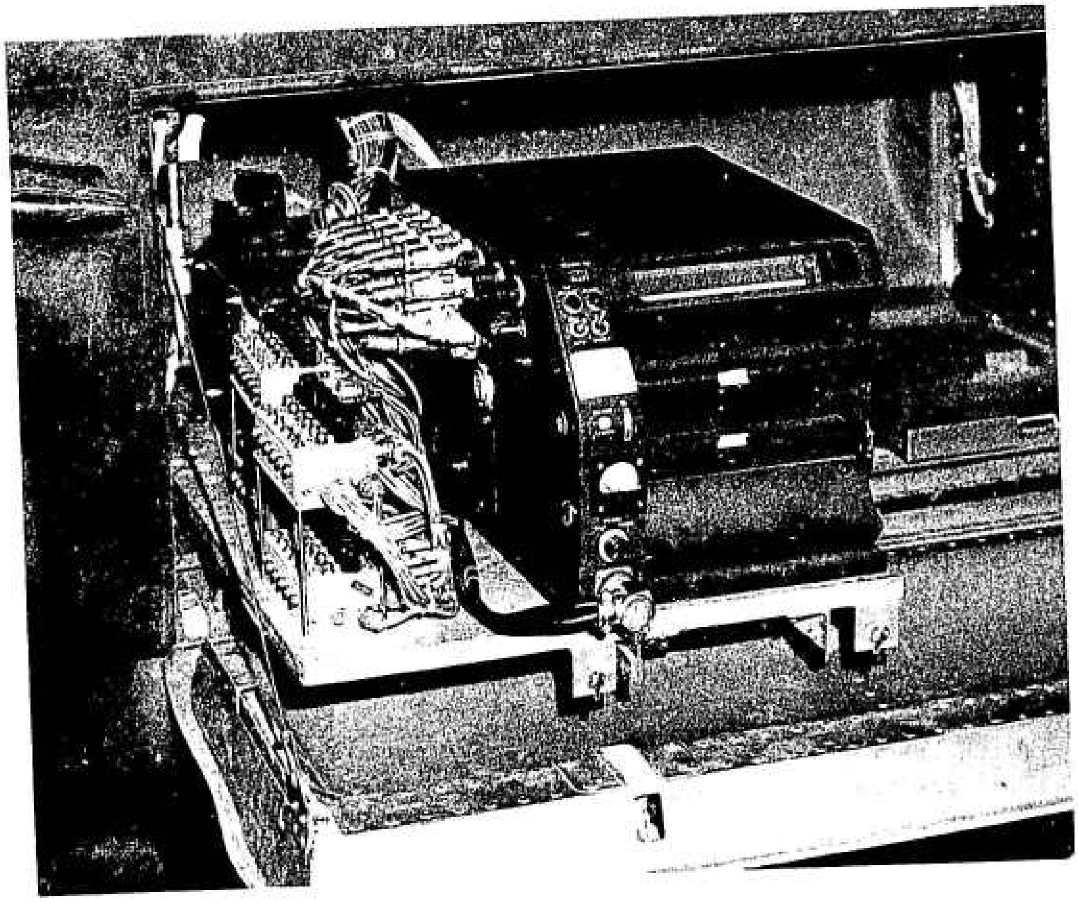


Photo 4. 24-Channel Oscillograph.

PHOTOPANEL

Boom system airspeed
Standard system airspeed
Boom system altimeter
Rotor speed
Gas producer speed
Fuel used total
Torque pressure (high)
Torque pressure (low)
Exhaust gas temperature
Compressor inlet temperature
Compressor inlet total pressure
Inlet guide vane position
Bleed band position (light)
Fuel pressure at nozzle
Time (10-second stopwatch)
Oscillograph correlation counter
Photopanel correlation counter
Engineer event
Pilot event

OSCILLOGRAPH

Longitudinal control position
Lateral control position
Directional control position
Collective control position
Pitch attitude
Roll attitude
Yaw attitude
Pitch rate
Roll rate
Yaw rate
CG (normal acceleration)
Angle of sideslip
Angle of attack
Engineer event
Pilot event
Photopanel correlation blip
Linear rotor speed
Gas producer speed
Inlet guide vane position
Bleed band position
Fuel pressure at nozzle
Tail rotor torque

APPENDIX V. TEST DATA

FIGURE 1

ACCELERATION PERFORMANCE

AH-1G T53-L-13

CLEAN CONFIGURATION

SYN	GRWT ~LB	DENS. ALT ~FT	LONG. C.G. ~INCH	ROTOR SPEED DURING ACCEL. ~RPM	AVG. ENGINE SHP DURING ACCEL ~SHP
○	8420	500	196.0 (MID)	324	1213
□	8390	500	196.0 (MID)	324	1200
◇	7340	500	195.9 (MID)	324	1202
△	7220	500	195.8 (MID)	324	1178

NOTE: DATA NOT CORRECTED FOR ENGINE POWER VARIATIONS

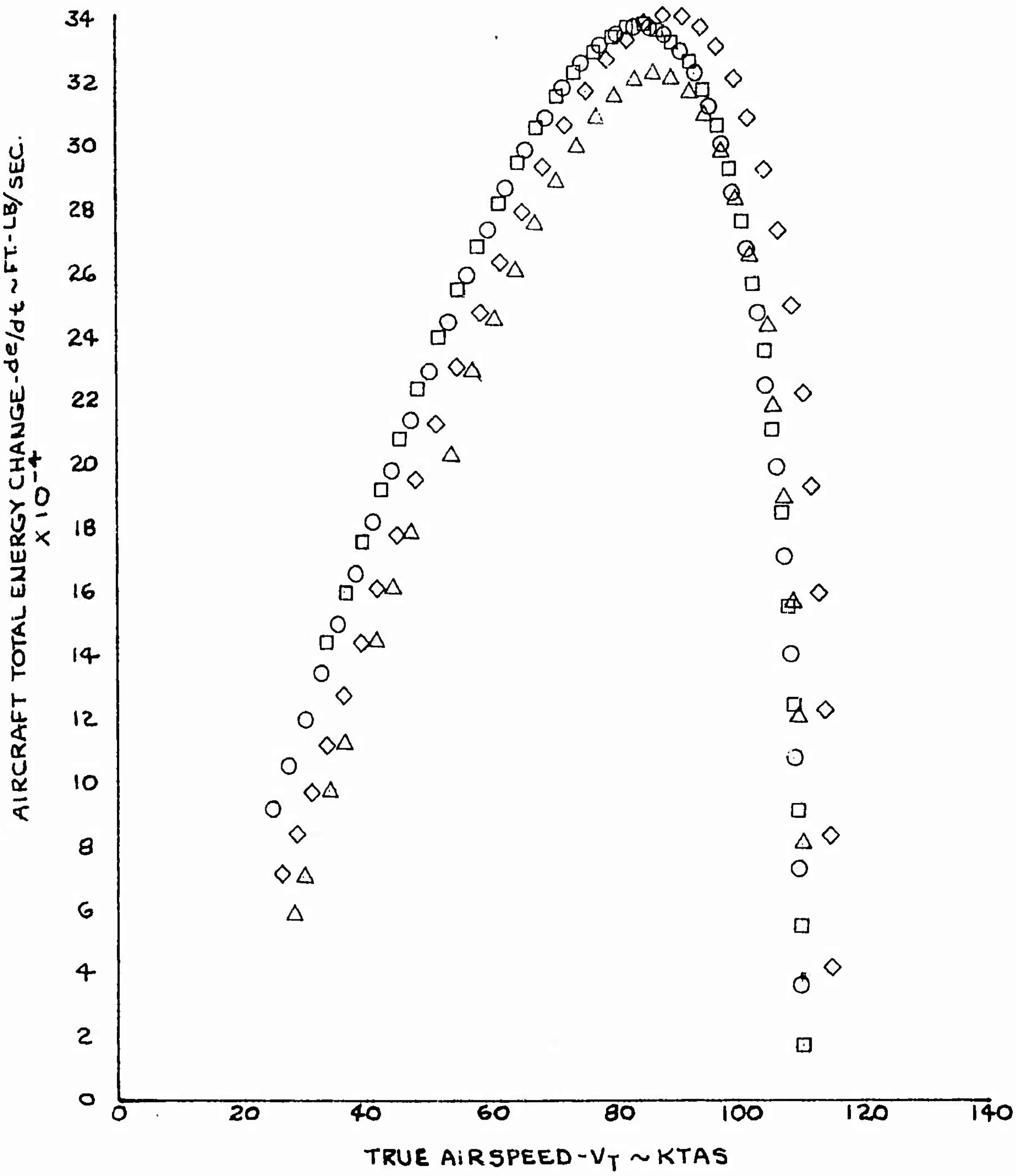


FIGURE 2

ACCELERATION PERFORMANCE

AH-1G T53-L-13

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SYM	GR WT ~LB.	DENS. ALT. H_D~FT.	LONG. C.G. ~INCH	ROTOR SPEED DURING ACCEL. ~RPM	AVG. ENGINE SHP DURING ACCEL. ~SHP
○	9485	500	195.9 (MID)	324	1178
□	9460	500	195.9 (MID)	324	1200
◇	8410	500	195.5 (MID)	324	1124
△	8390	500	195.5 (MID)	324	1188

NOTE: DATA NOT CORRECTED FOR ENGINE POWER VARIATIONS

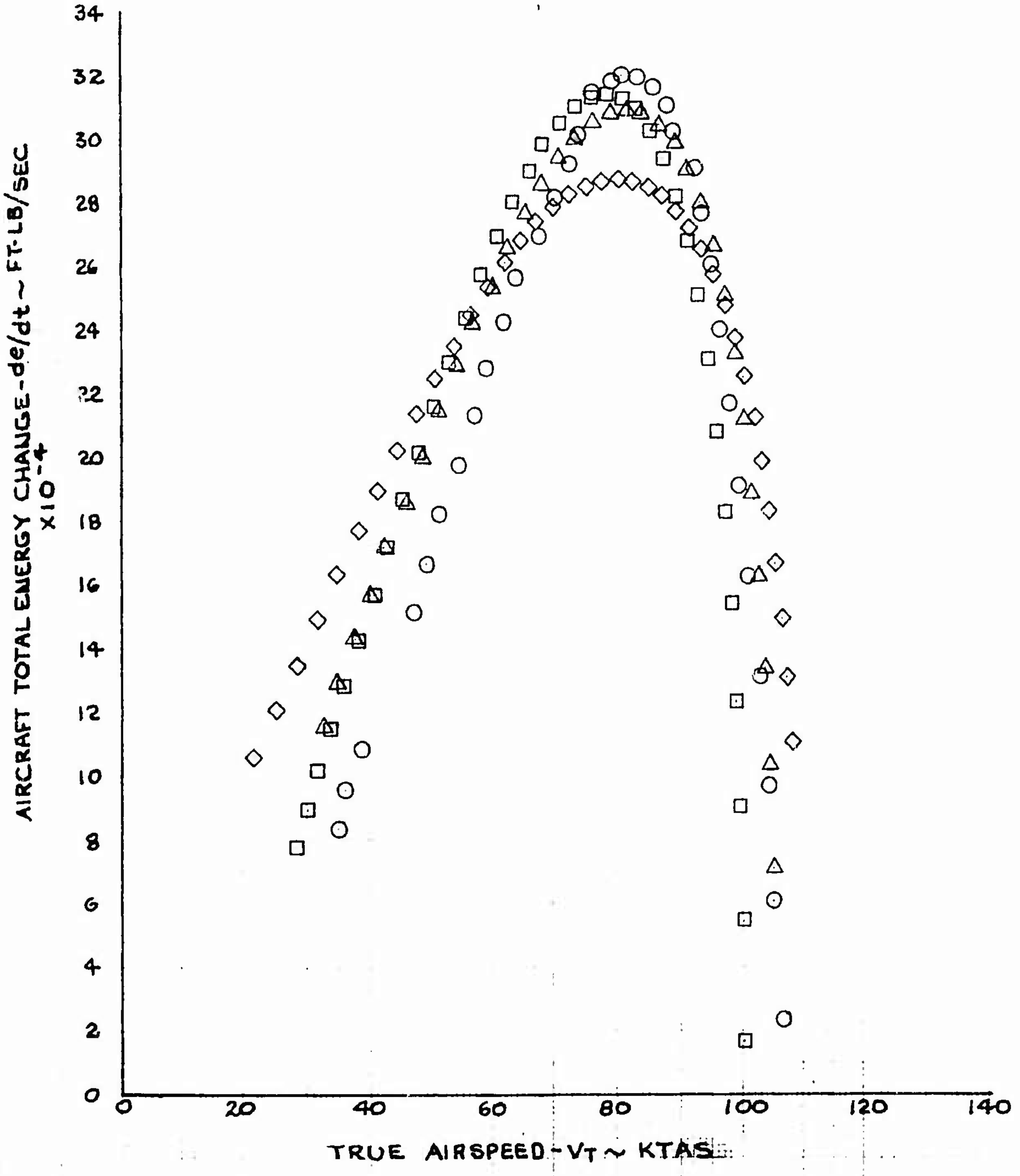


FIGURE 3
ACCELERATION PERFORMANCE
AH-1G T53-L-13
HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SYM	GRWT ~LB	DENS. ALT. H_D ~ FT.	LONG. C.G. ~INCH	ROTOR SPEED DURING ACCEL. ~RPM	AVG. ENGINE SHP DURING ACCEL. ~SHP
○	9160	500	195.9(MID)	324	1200
□	8410	500	195.5(MID)	324	1124

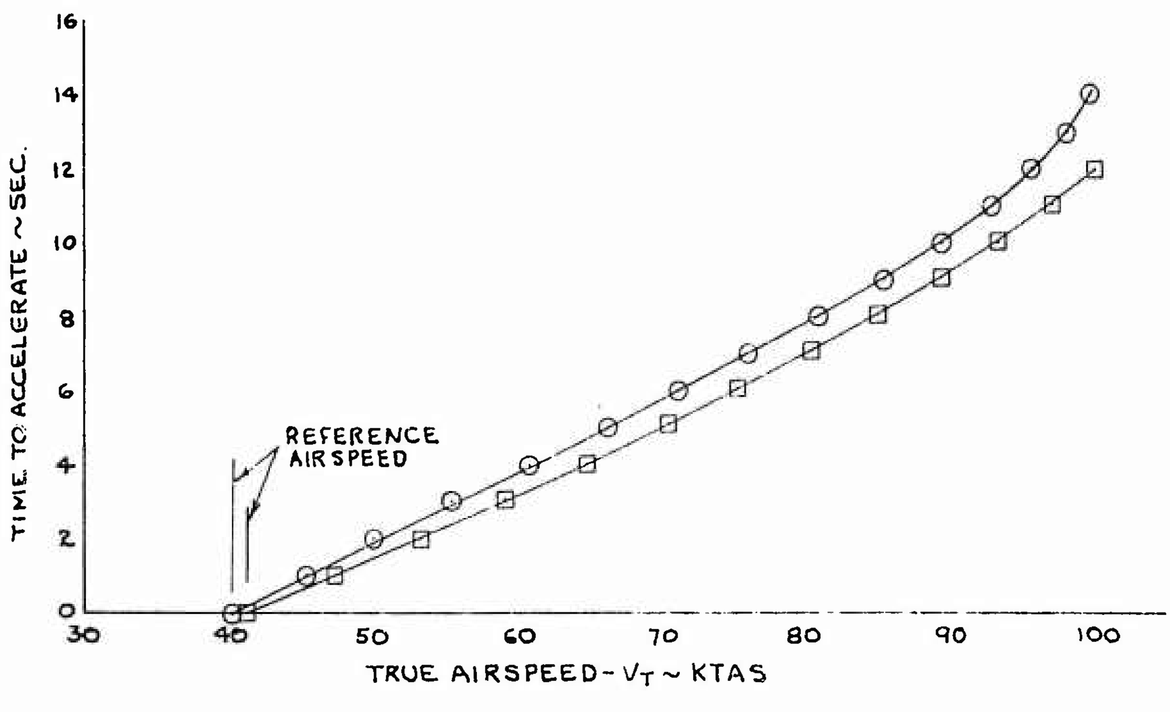


FIGURE 4

DECELERATION PERFORMANCE

AH-1G T53-L-13

SYM	GRWT ~ LB	DENS. ALT. H_D ~ FT.	LONG. C.G. ~ INCH	ROTOR SPEED AT START OF DECEL ~ RPM	ARMAMENT CONFIG.
○	8470	500	195.5 (MID)	324	HVY. HOG
□	9350	500	195.8 (MID)	324	HVY. HOG
◇	8330	500	196.0 (MID)	324	CLEAN
△	7170	500	195.8 (MID)	324	CLEAN

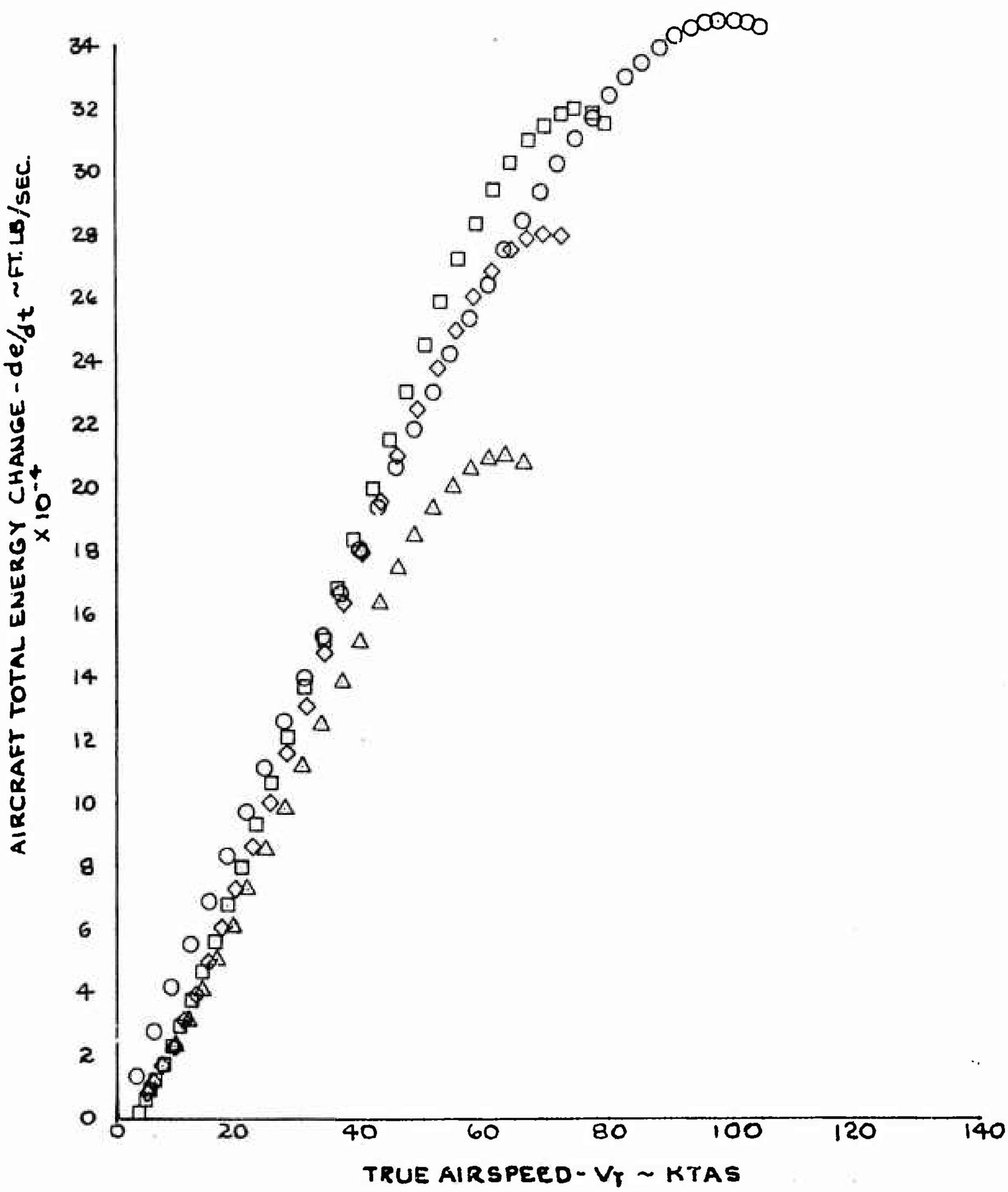


FIGURE 5

DECELERATION PERFORMANCE

AH-1G T53-L-13

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SYM	GRWT ~ LB	DENS. ALT. H_D ~ FT.	LONG. C.G. ~ INCH	ROTOR SPEED AT START OF DECEL. ~ RPM
O	9260	500	195.8 (MID)	324
□	8470	500	195.5 (MID)	324

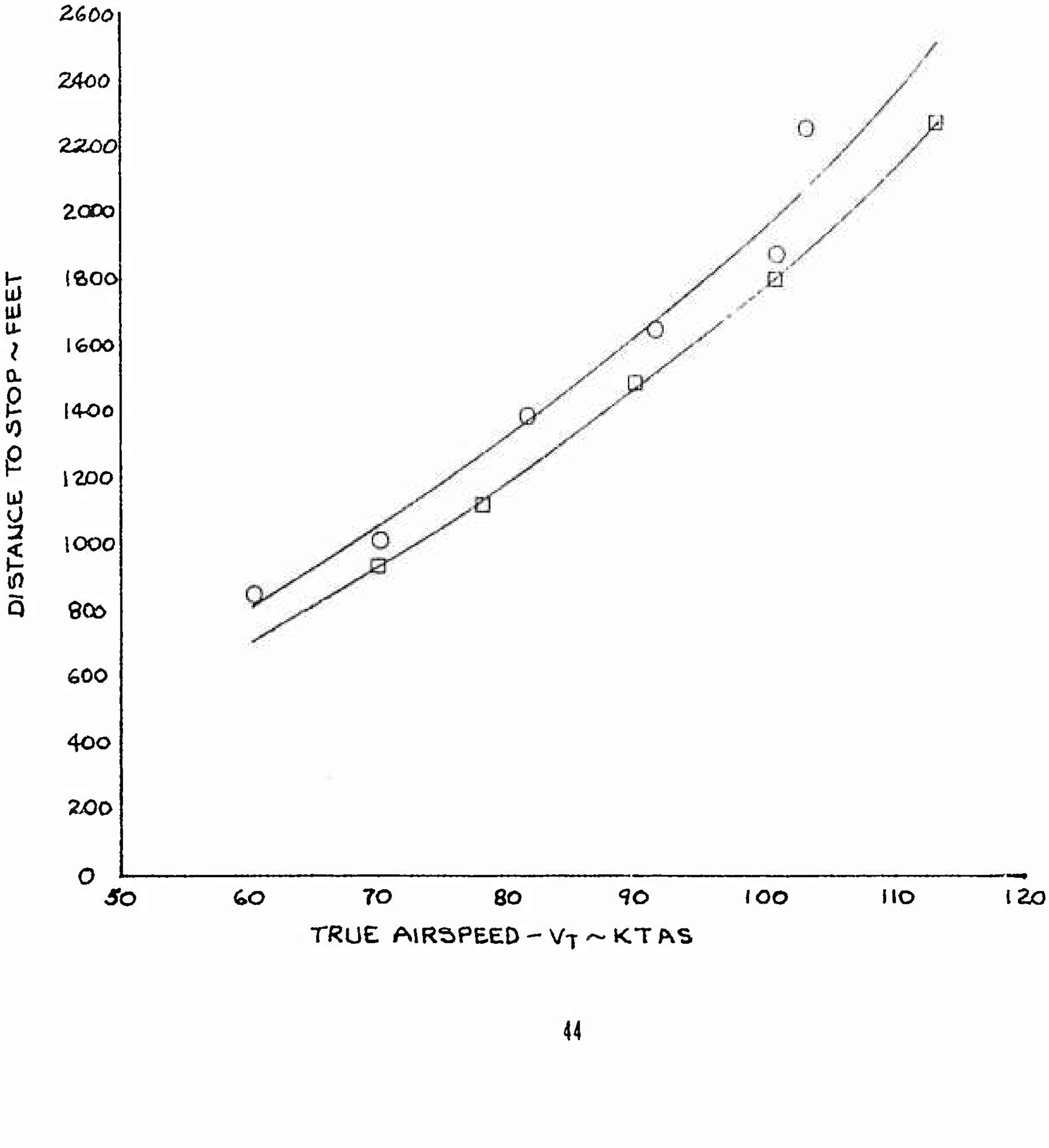


FIGURE 6

DECELERATION PERFORMANCE

AH-1G T53-L-13

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SYM.	GRWT ~LB	DENS. ALT. H_D ~ FT.	LONG. C.G. ~INCH	ROTOR SPEED AT START OF DECEL. ~ RPM
O	9260	500	195.8 (MID)	324
□	8470	500	195.5 (MID)	324

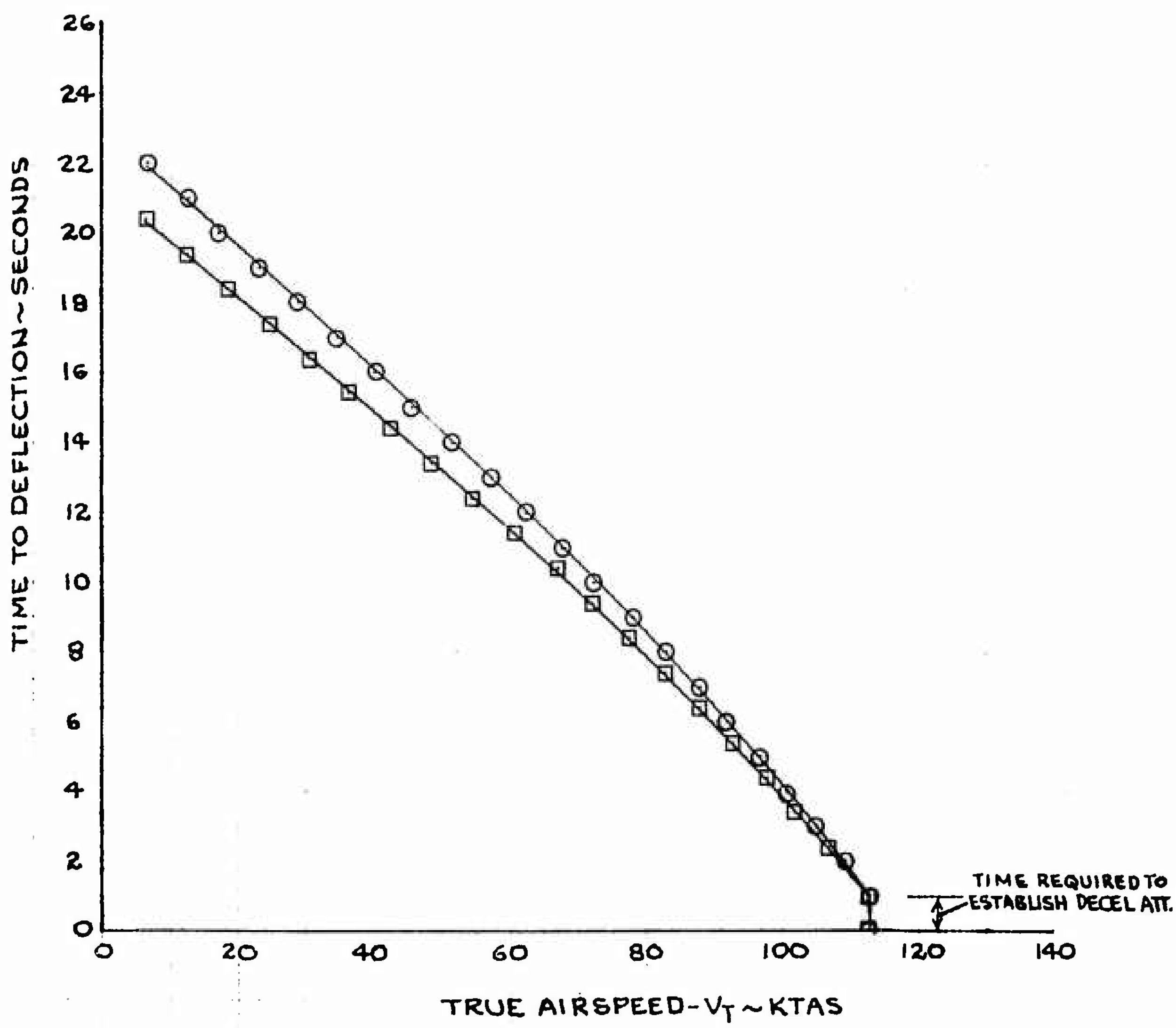


FIGURE 7

ACCELERATION PERFORMANCE

AH-1G USA S/N 66-15247

CLEAN CONFIGURATION

GRWT ~LB	DENSITY ALTITUDE H_D ~FT	LONG. C.G. POSITION ~INCHES	ROTOR SPEED ~RPM
7550	270	196.1 (MID)	323.5

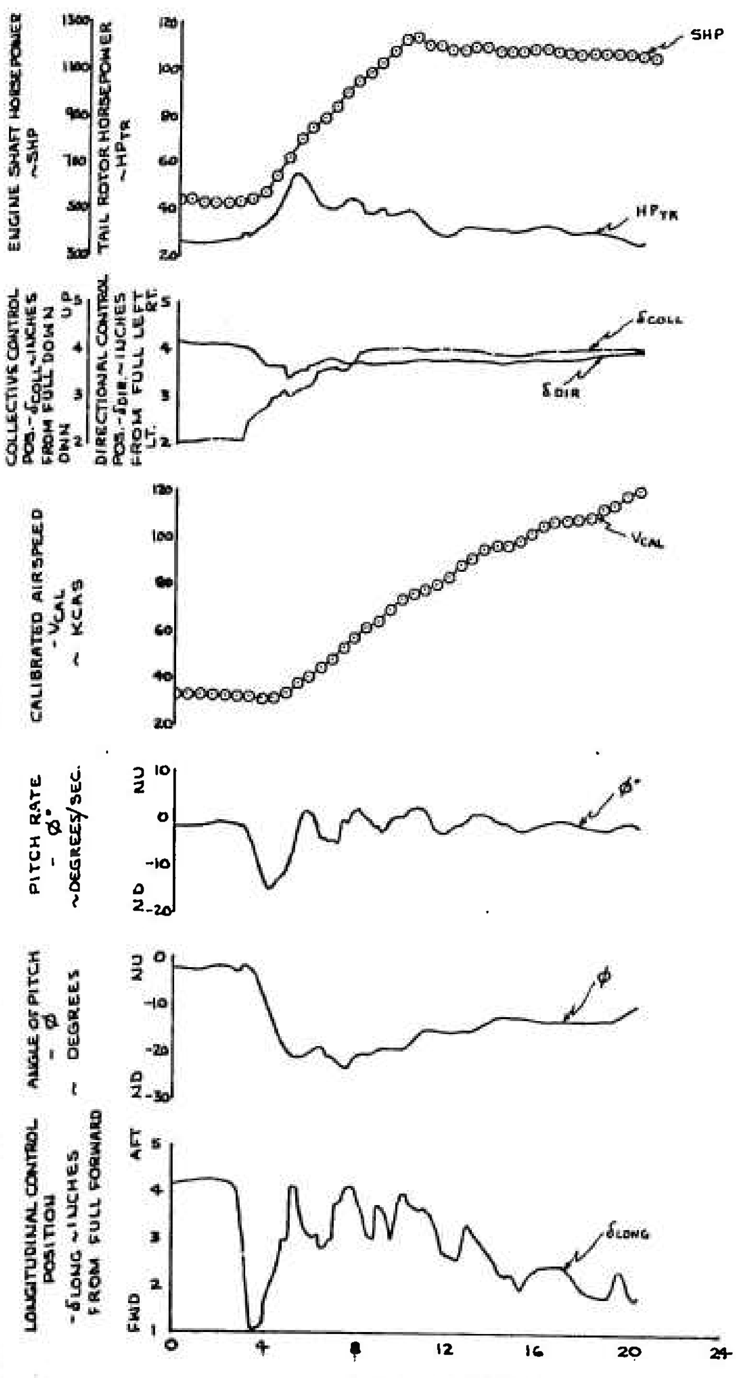


FIGURE 8

ACCELERATION PERFORMANCE

AH-1G USA S/N 615247

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

GRWT ~LB	DENSITY ALTITUDE H_D ~ FEET	LONG. C.G. POSITION ~INCHES	ROTOR SPEED ~RPM
9460	310	195.9(MID)	323.0

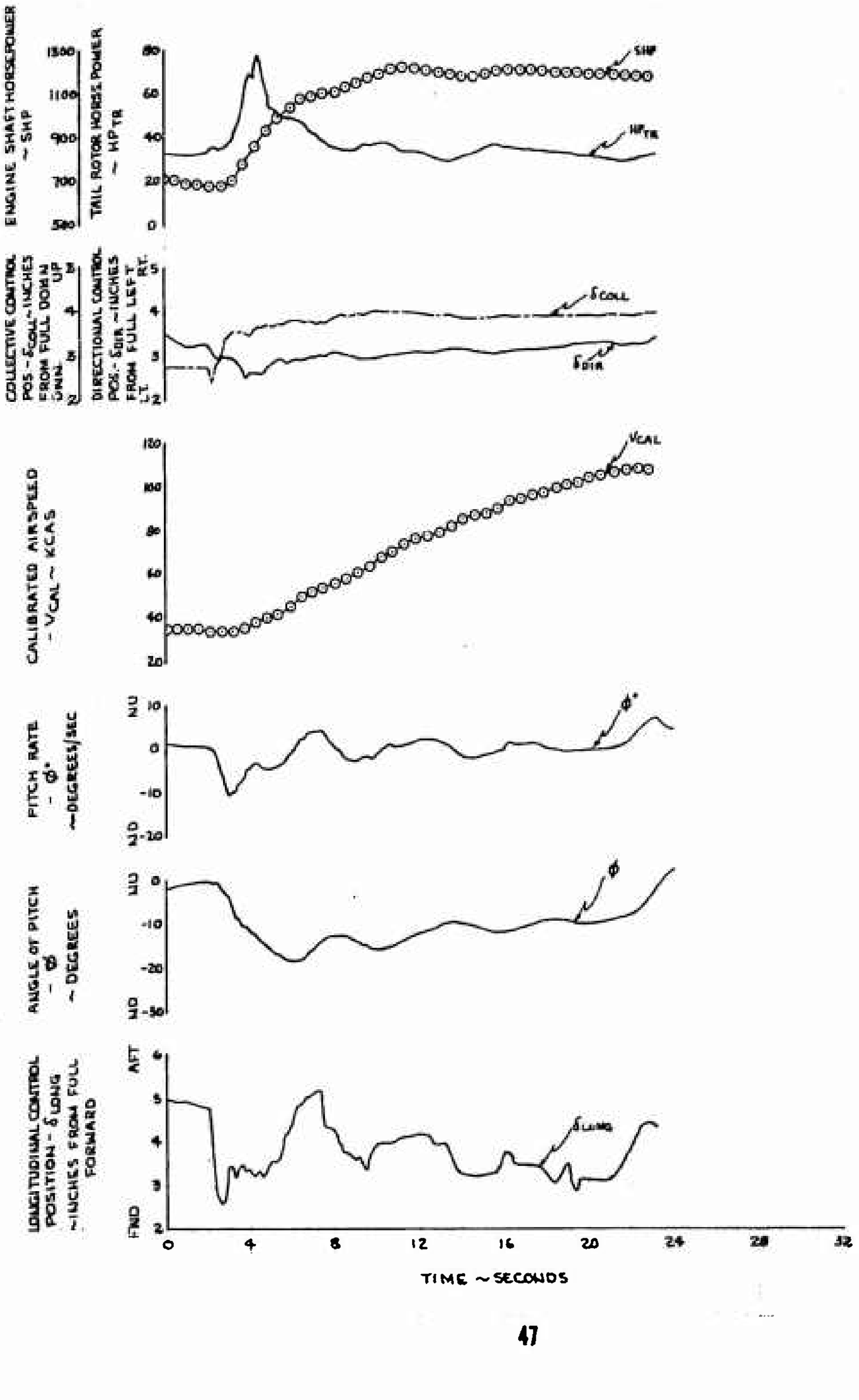


FIGURE 9
DECELERATION PERFORMANCE
AH-1G USA S/N 615247
CLEAN CONFIGURATION

GRWT ~LB	DENSITY ALTITUDE H_D ~ FEET	LONG. C.G. POSITION ~INCHES
7465	280	196.1 (MID)

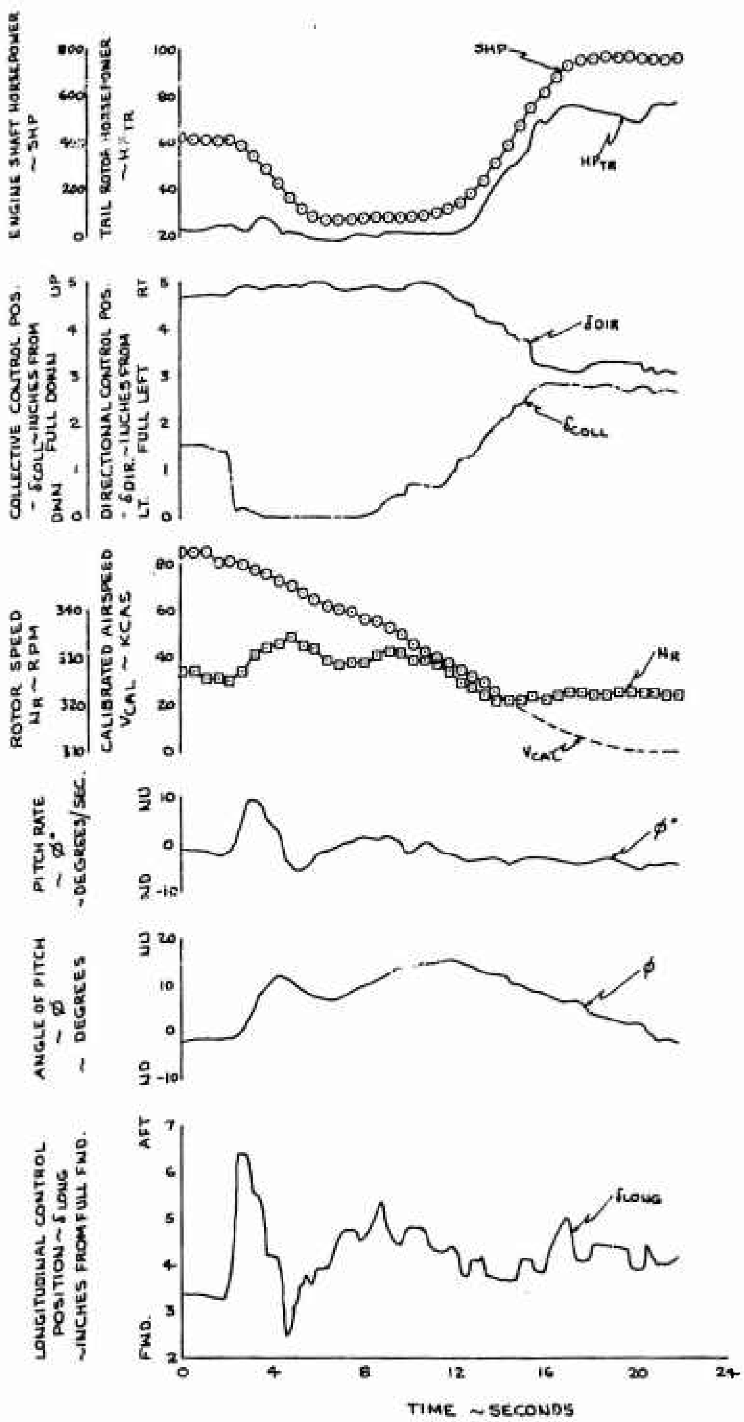


FIGURE 10
DECELERATION PERFORMANCE
AH-1G USA S/N 615247
HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

GRWT ~LB	DENSITY ALTITUDE H_D ~ FEET	LONG. C.G. POSITION ~INCHES
9350	320	195.7(MID)

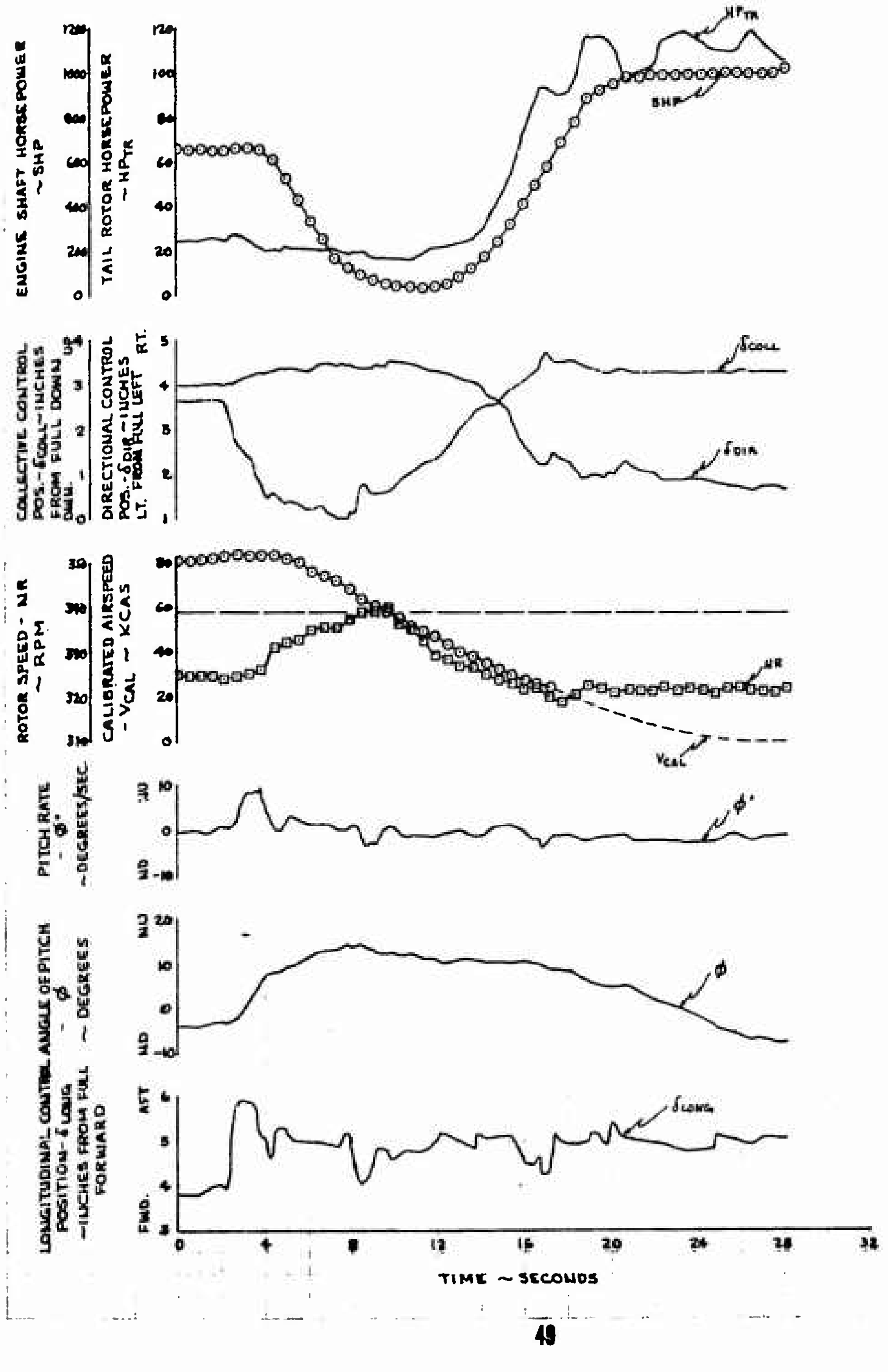


FIGURE 11

TEAR DROP TURNING PERFORMANCE

AH-1G T53-L-13

CLEAN CONFIGURATION

AVG. GRWT ~LB	AVG. ALT. H_D ~FT.	AVG. LONG. C.G. ~INCH	ROTOR SPEED ~RPM
7700	3100	196.0(MID)	324

NOTES: 1. SHADED SYMBOLS DENOTE LEFT TURN
2. OPEN SYMBOLS DENOTE RIGHT TURN

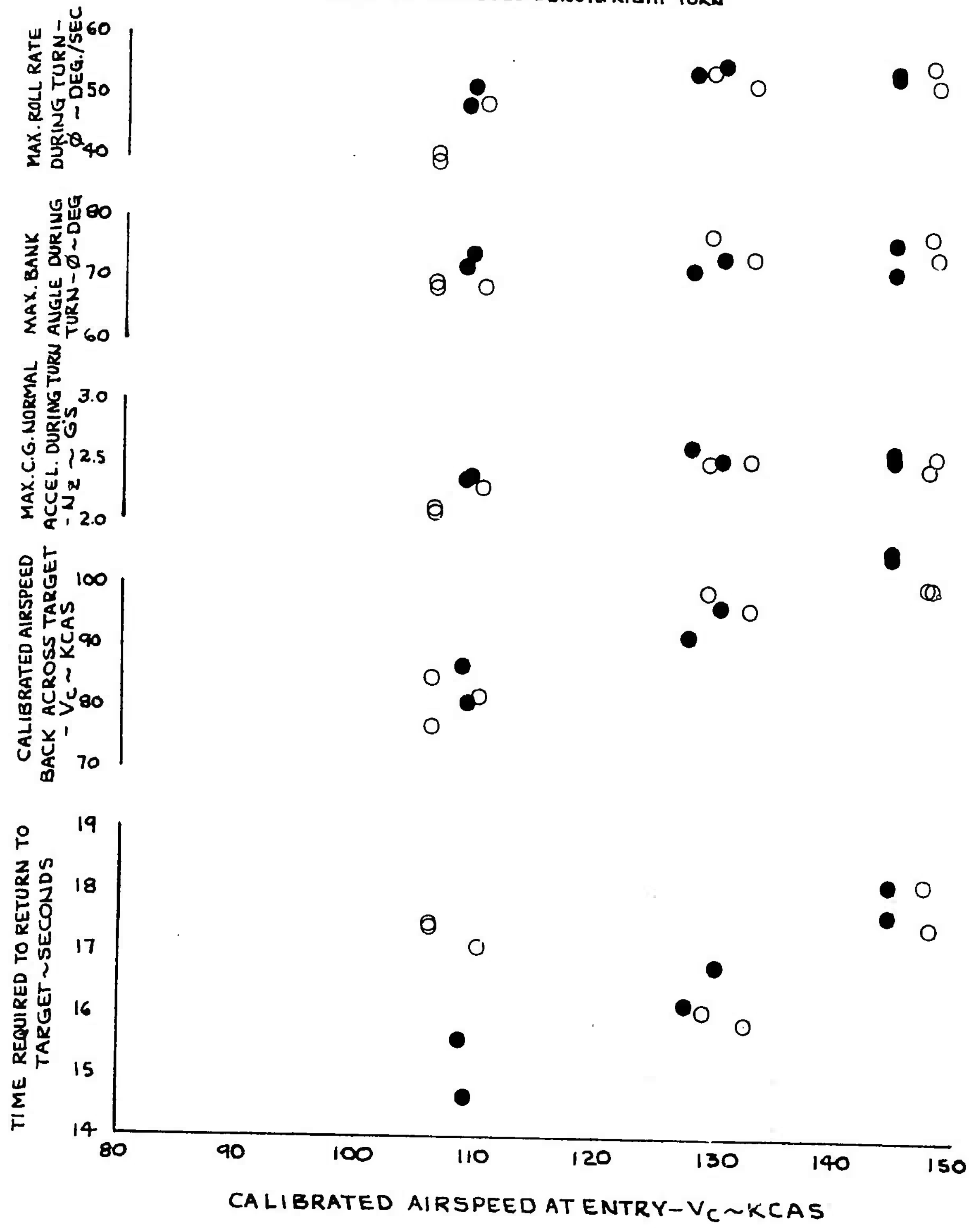


FIGURE 12

TEAR DROP TURNING PERFORMANCE

AH-1G T53-L-13

CLEAN CONFIGURATION

AVG. GRWT ~LB	AVG. ALT. H_D ~ FT.	AVG. LONG. C.G. ~ INCH	ROTOR SPEED ~ RPM
8600	2400	193.5(MID)	324

NOTES: 1. SHADED SYMBOLS DENOTE LEFT TURN
2. OPEN SYMBOLS DENOTE RIGHT TURN

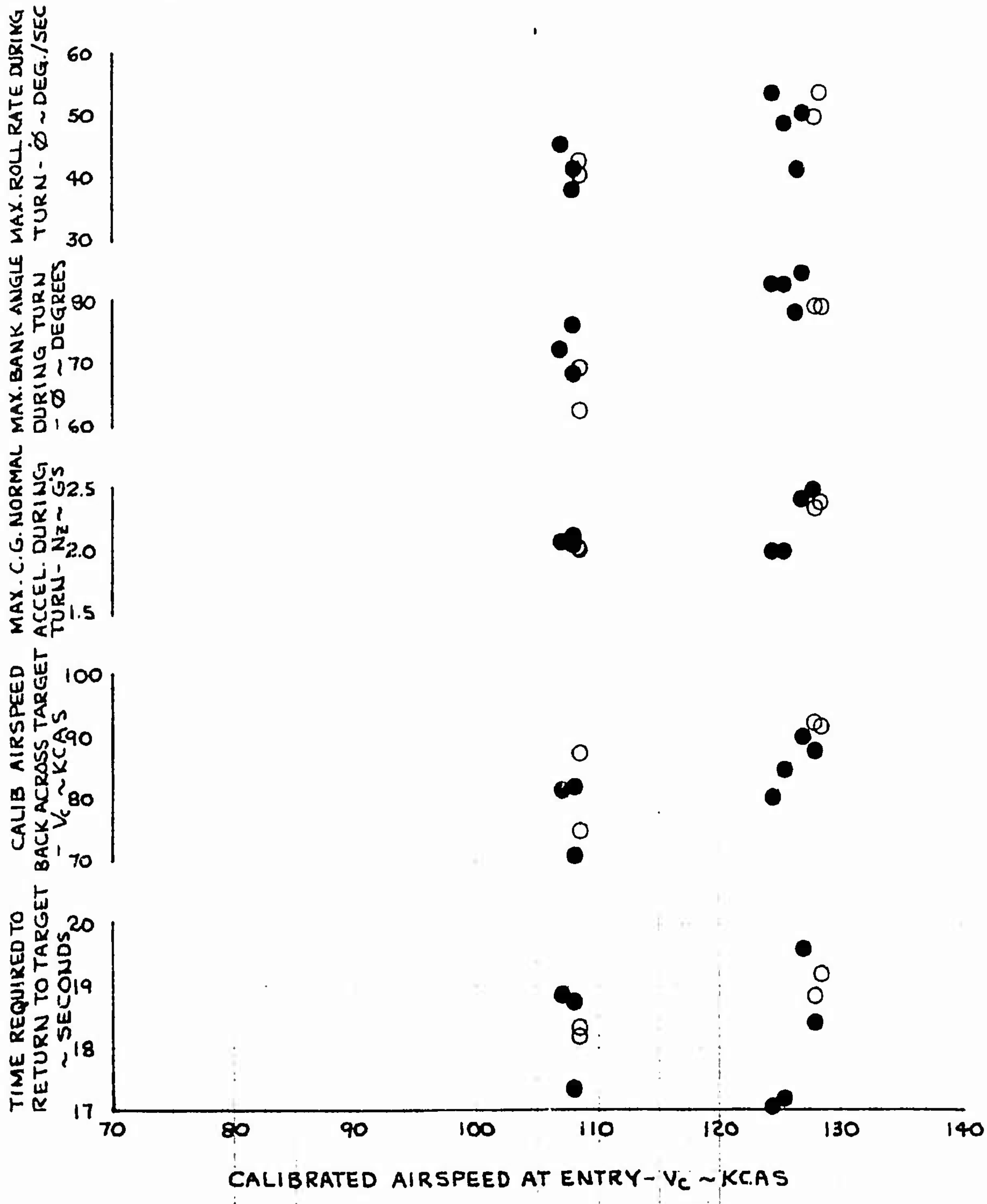


FIGURE 13
TEAR DROP TURNING PERFORMANCE
AH-1G T53-L-13
HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

AVG. GRWT ~ LB	AVG. ALT. H_D ~ FT	AVG. LONG. C.G. ~ INCH	ROTOR SPEED ~ RPM
8520	2400	195.9 (MID)	324

NOTES: 1. SHADED SYMBOLS DENOTE LEFT TURN
2. OPEN SYMBOLS DENOTE RIGHT TURN

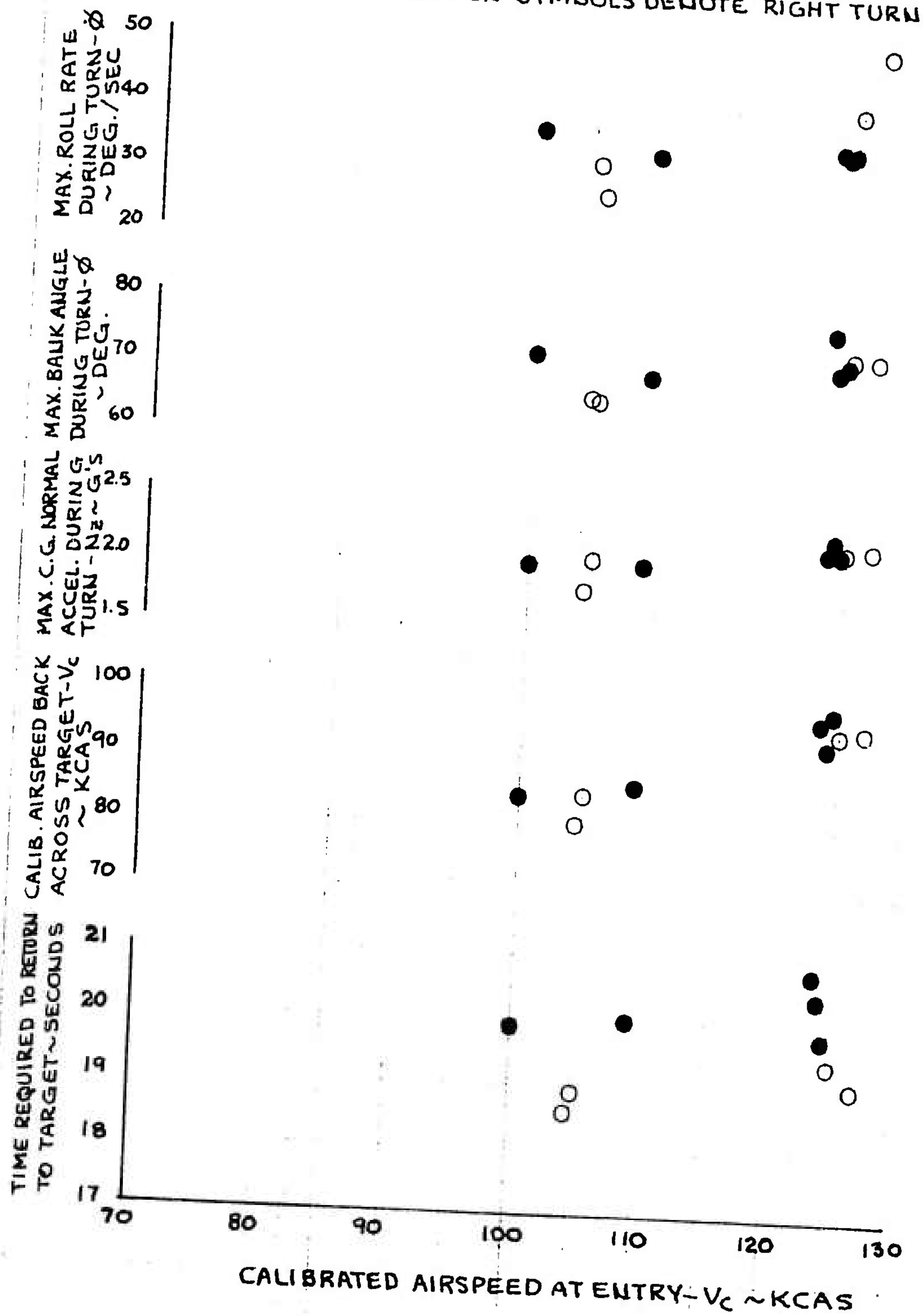


FIGURE 14

TEAR DROP TURNING PERFORMANCE

AH-1G T53-L-13

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

AVG. GRWT ~ LB	AVG. ALT. H_D ~ FT	AVG. LONG. C.G. ~ INCH	ROTOR SPEED ~ RPM
9400	3000	196.3 (MID)	324

NOTES: 1. SHADED SYMBOLS DENOTE LEFT TURN
2. OPEN SYMBOLS DENOTE RIGHT TURN

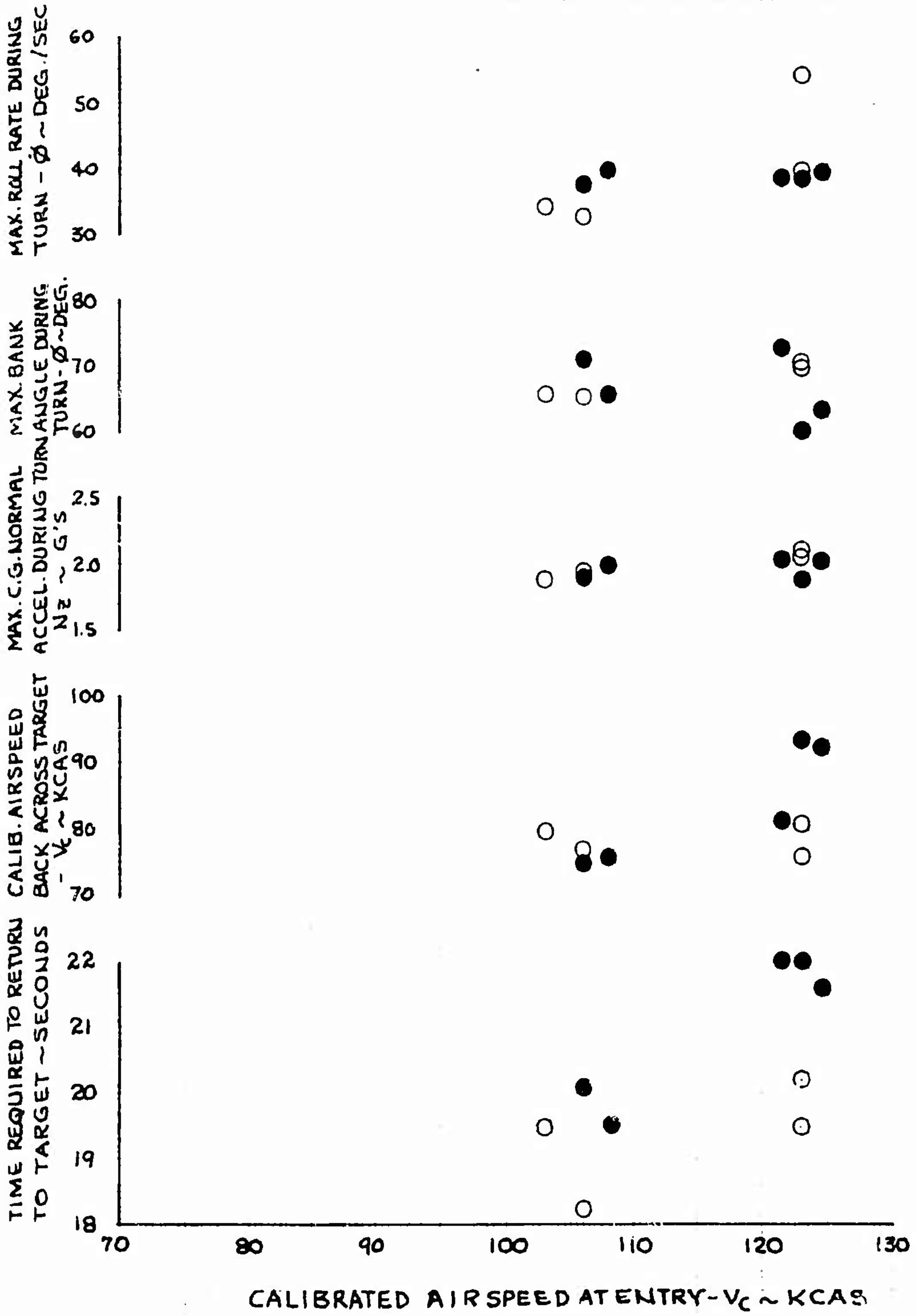


FIGURE 15.

180 DEGREE TURNING PERFORMANCE

AH-1G T53-L-13

CLEAN CONFIGURATION

AVG. GRWT. ~LB	AVG. ALT. H_D~FT.	AVG. LONG. C.G. ~INCH	ROTOR SPEED ~RPM
7250	3100	195.6(MID)	324

NOTES: 1. DATA PRESENTED FOR ZERO WIND CONDITION

2. DATA BASED ON ZERO ALTITUDE LOSS DURING TURN

3. DATA REPRESENTS TURNING PERFORMANCE FOR BOTH LEFT & RIGHT TURN

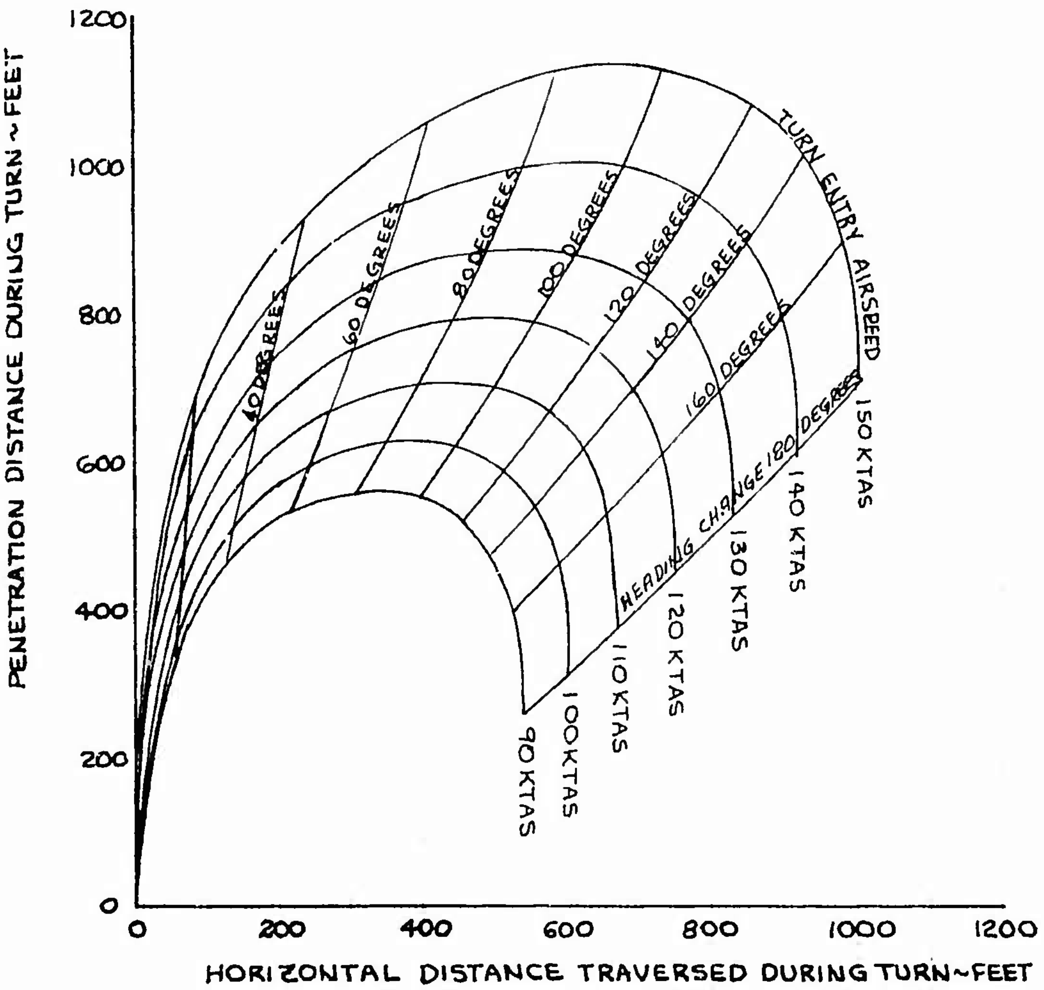


FIGURE 16

180 DEGREE TURNING PERFORMANCE

AH-1G T53-L-13

CLEAN CONFIGURATION

AVG. GRWT ~LB	AVG. ALT. H_D ~FT.	AVG. LONG. C.G ~INCH	ROTOR SPEED ~RPM
8250	2400	193.1 (MID)	324

NOTES: 1. DATA PRESENTED FOR ZERO WIND CONDITION
2. DATA BASED ON ZERO ALTITUDE LOSS DURING TURN
3. DATA REPRESENTS TURNING PERFORMANCE FOR BOTH LEFT & RIGHT TURN

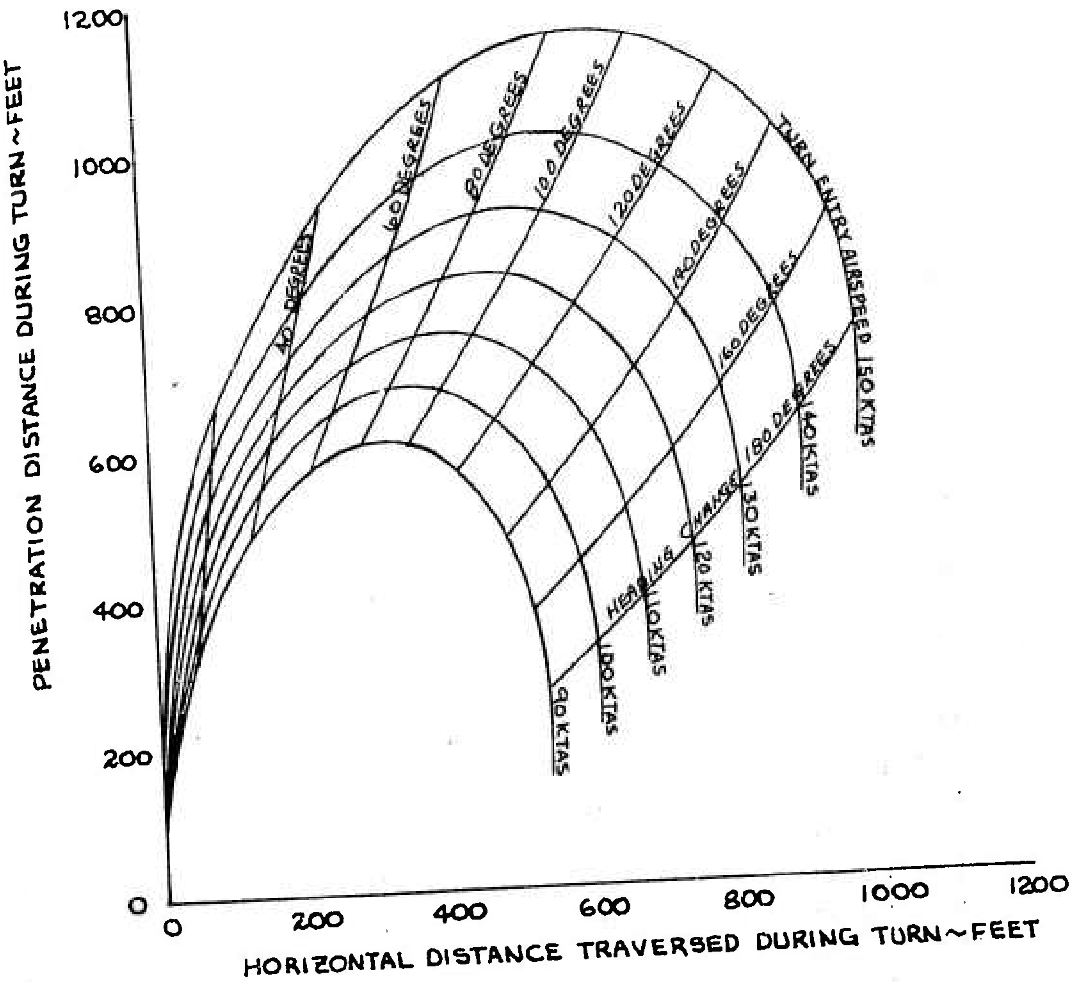


Figure 17

180 Degree Turning Performance

AH-1G T53-L-13

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

AVG. GRWT. ~LB.	AVG. ALT. H_D~FT.	AVG. LONG. C.G. ~INCH	ROTOR SPEED ~RPM
8250	2400	195.6 (MID)	324

NOTES: 1. DATA PRESENTED FOR ZERO WIND CONDITION
2. DATA BASED ON ZERO ALTITUDE LOSS DURING TURN
3. DATA REPRESENTS TURNING PERFORMANCE FOR BOTH LEFT AND RIGHT TURNS

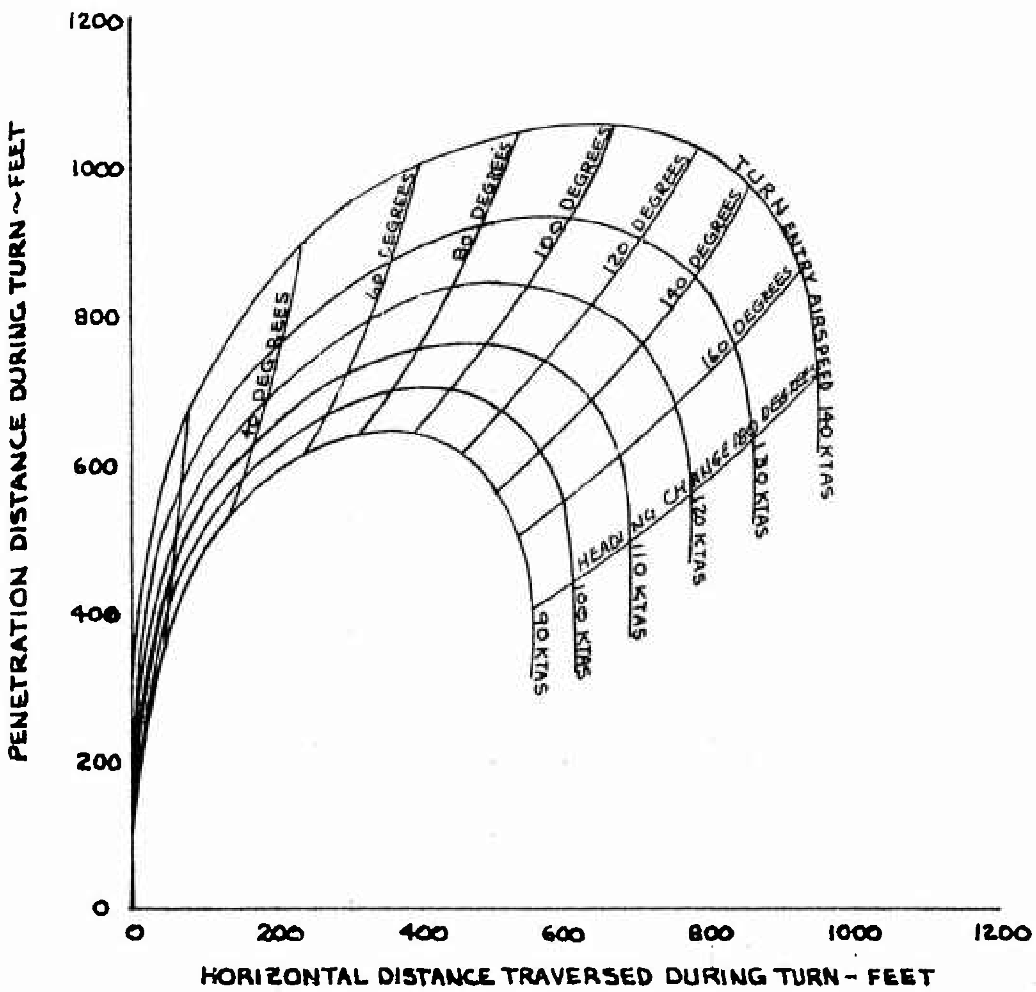


FIGURE 18

180 DEGREE TURNING PERFORMANCE

AH-1G T53-L-13

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

AVG. GRWT ~ LB	AVG. ALT H_D ~ FT.	AVG. LONG C.G. ~ INCH	ROTOR SPEED ~ RPM
9100	3000	196.2 (MID)	324

NOTES: 1. DATA PRESENTED FOR ZERO WIND CONDITION

2. DATA BASED ON ZERO ALTITUDE LOSS DURING TURN

3. DATA REPRESENTS TURNING PERFORMANCE FOR BOTH LEFT & RIGHT TURN

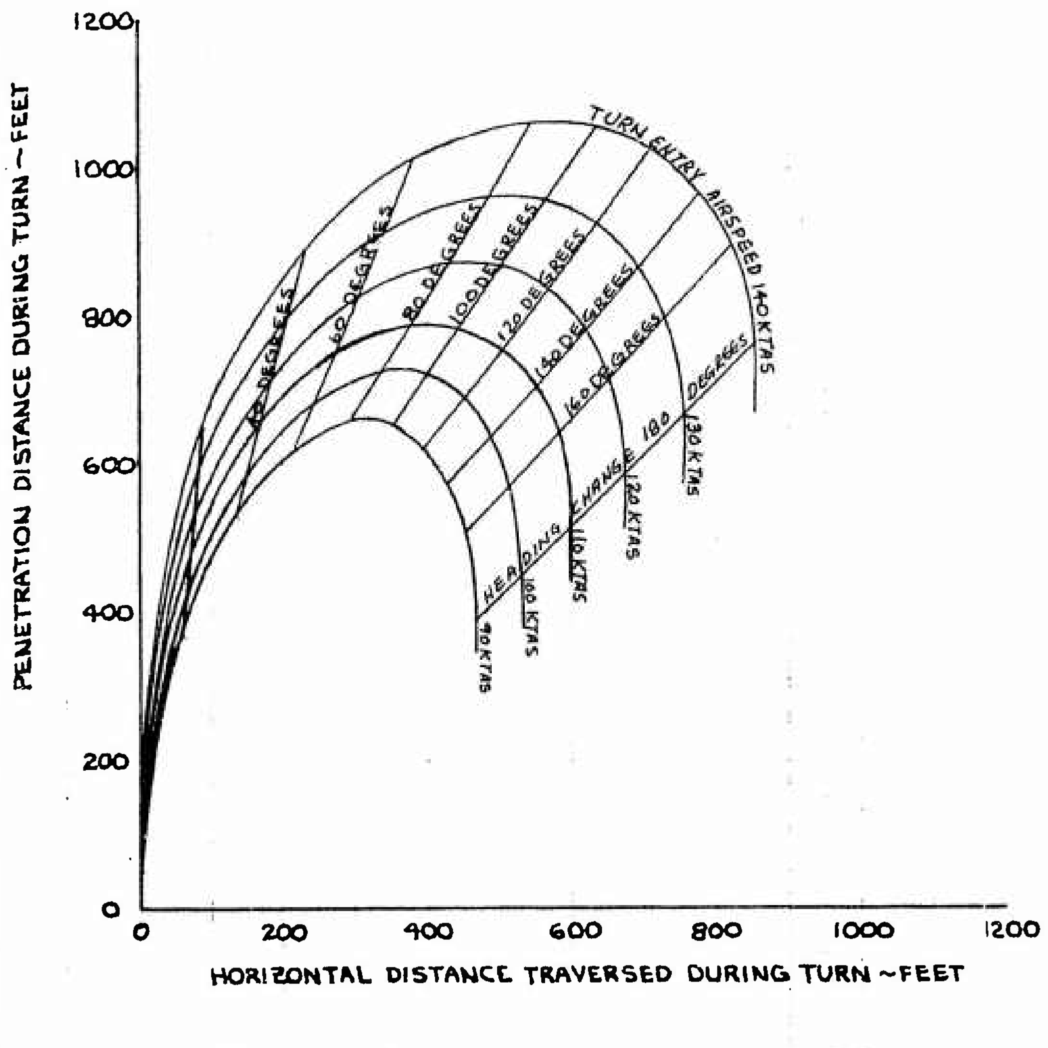


FIGURE 19
180 DEGREE TURNING PERFORMANCE
AH-1G USA S/N 615247
CLEAN CONFIGURATION

AVG. GRWT ~ LB.	AVG. ALT. H_D ~ FT.	AVG. LONG. C.G. ~ INCH	ROTOR SPEED ~ RPM
7250	3100	195.6 (MID)	324

NOTES: 1. CIRCULAR SYMBOLS DENOTE RIGHT TURN
2. SQUARE SYMBOLS DENOTE LEFT TURN
3. ALL TURNS WERE INITIATED FROM LEVEL FLIGHT

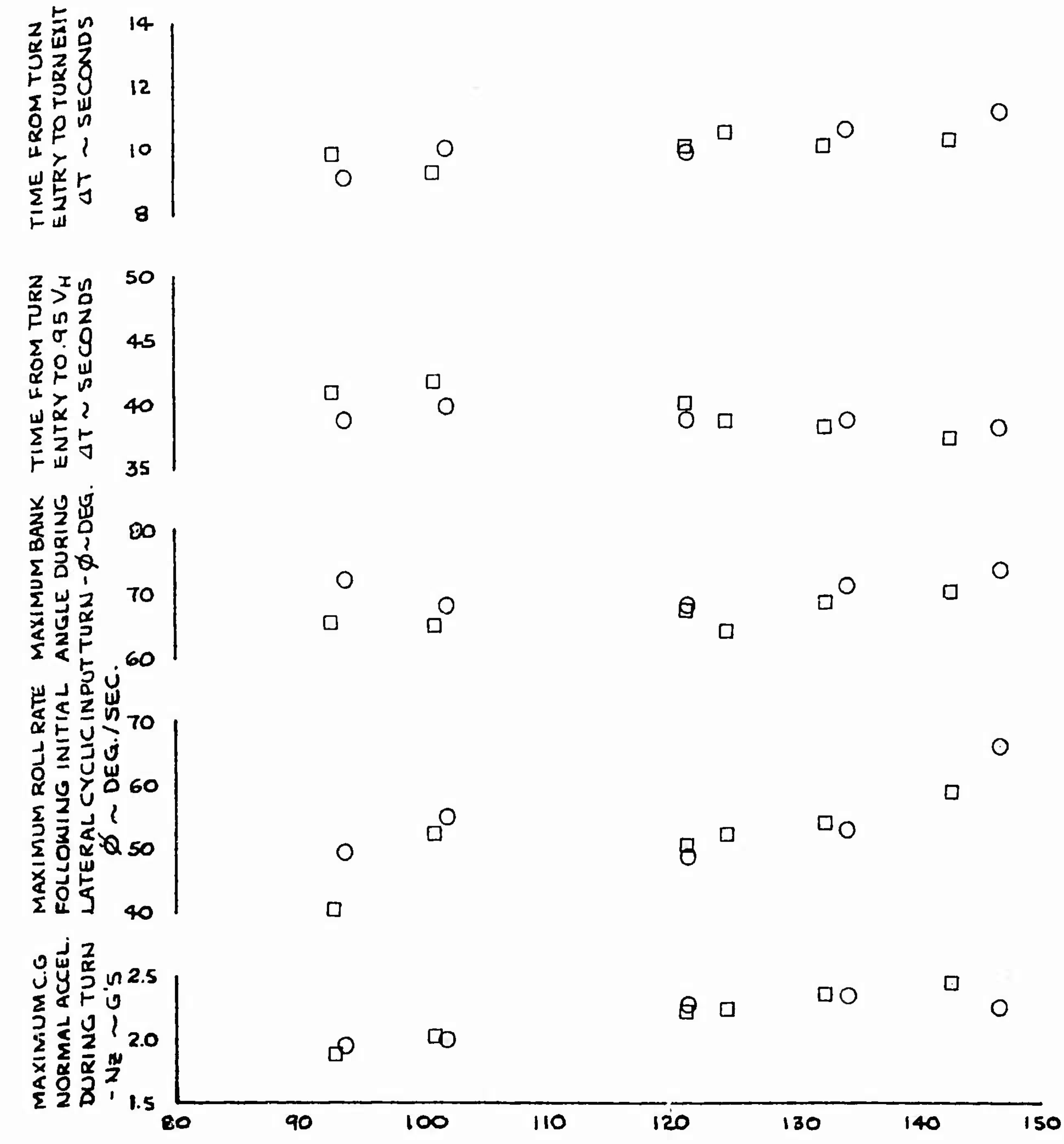


FIGURE 20

180 DEGREE TURNING PERFORMANCE

AH-1G USA S/N 615247

CLEAN CONFIGURATION

AVG. GRWT ~LB	AVG. ALT. H_D~FT	AVG. LONG. C.G. ~INCH	ROTOR SPEED ~ RPM
8250	2400	193.1 (MID)	324

NOTES: 1. CIRCULAR SYMBOLS DENOTE RIGHT TURN
2. SQUARE SYMBOLS DENOTE LEFT TURN
3. ALL TURNS WERE INITIATED FROM LEVEL FLIGHT

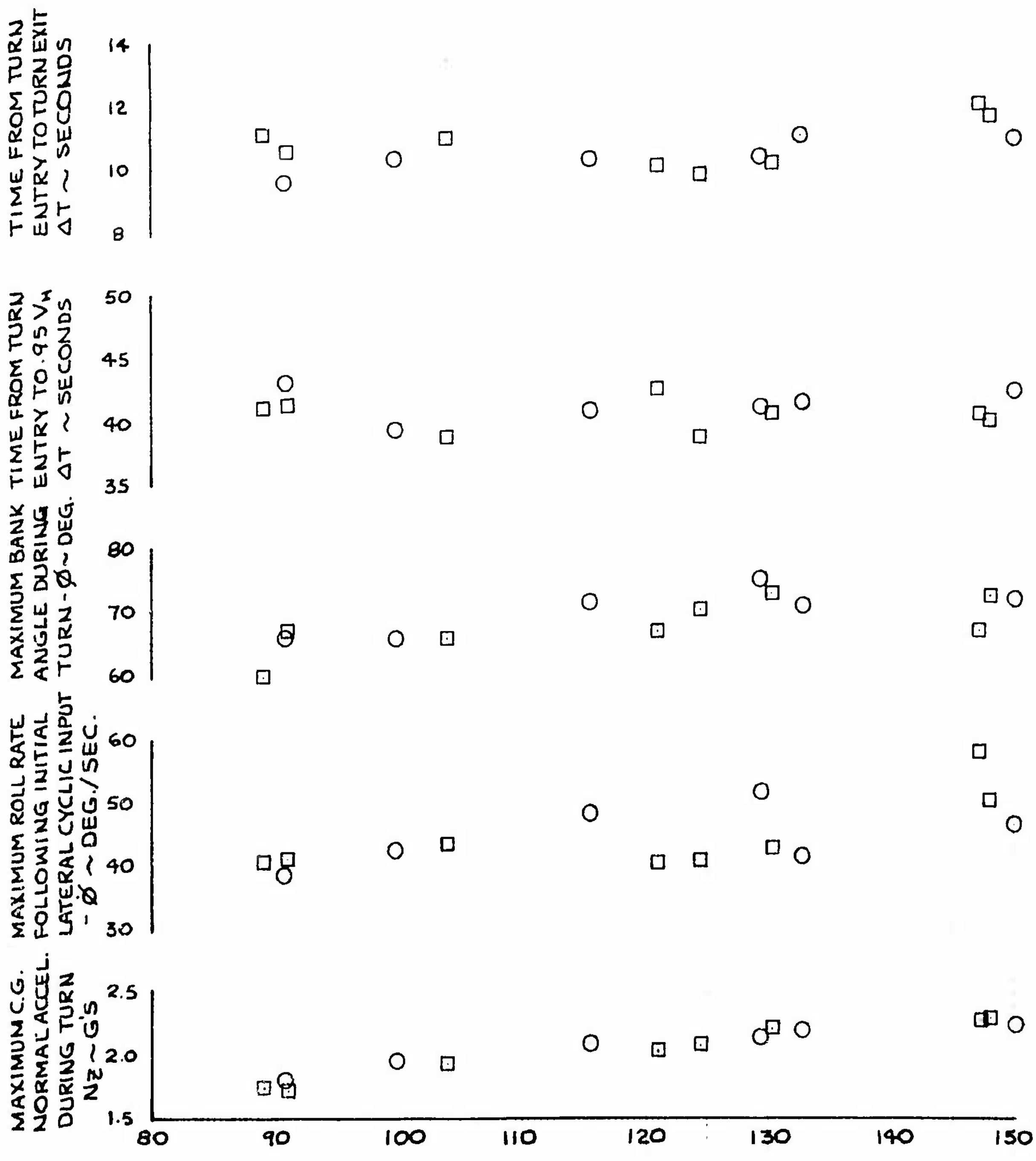


FIGURE 21

180 DEGREE TURNING PERFORMANCE

AH-1G USA S/N 615247

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

AVG. GRWT ~LB	AVG. ALT. H_D ~ FT.	AVG. LONG. C.G. ~ INCH	ROTOR SPEED ~ RPM
8250	2500	195.6 (MID)	324

NOTES: 1. CIRCULAR SYMBOLS DENOTE RIGHT TURN
2. SQUARE SYMBOLS DENOTE LEFT TURN
3. ALL TURNS WERE INITIATED FROM LEVEL FLIGHT

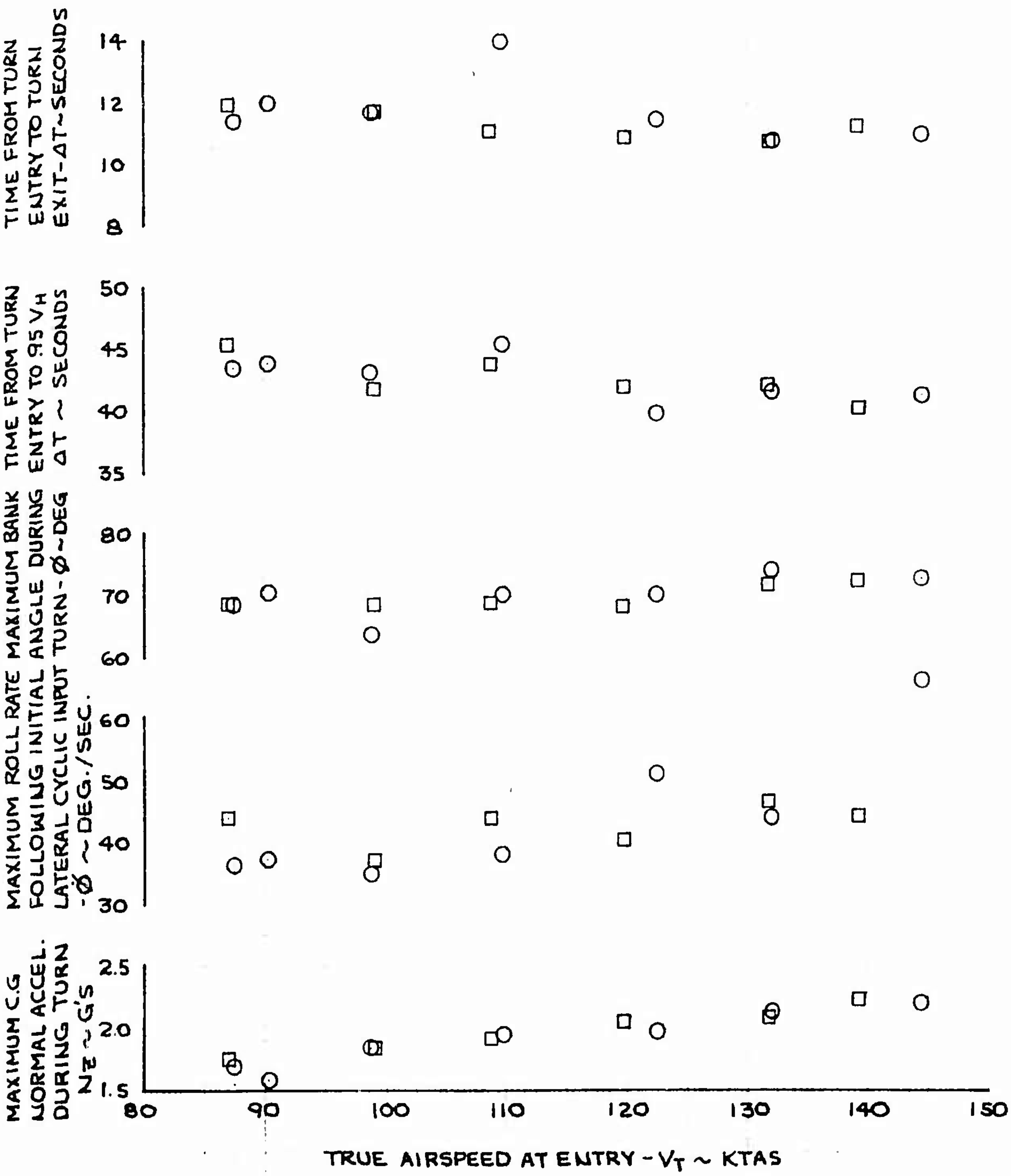


FIGURE 22

180 DEGREE TURNING PERFORMANCE

AH-1G USA S/N 615247

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

AVG. GRWT ~LB	AVG ALT. H_D ~ FT	AVG. LONG. C.G. ~INCH	ROTOR SPEED ~ RPM
9100	3000	196.2(MID)	324

NOTES: 1. CIRCULAR SYMBOLS DENOTE RIGHT TURN
2. SQUARE SYMBOLS DENOTE LEFT TURN
3. ALL TURNS WERE INITIATED FROM LEVEL FLIGHT

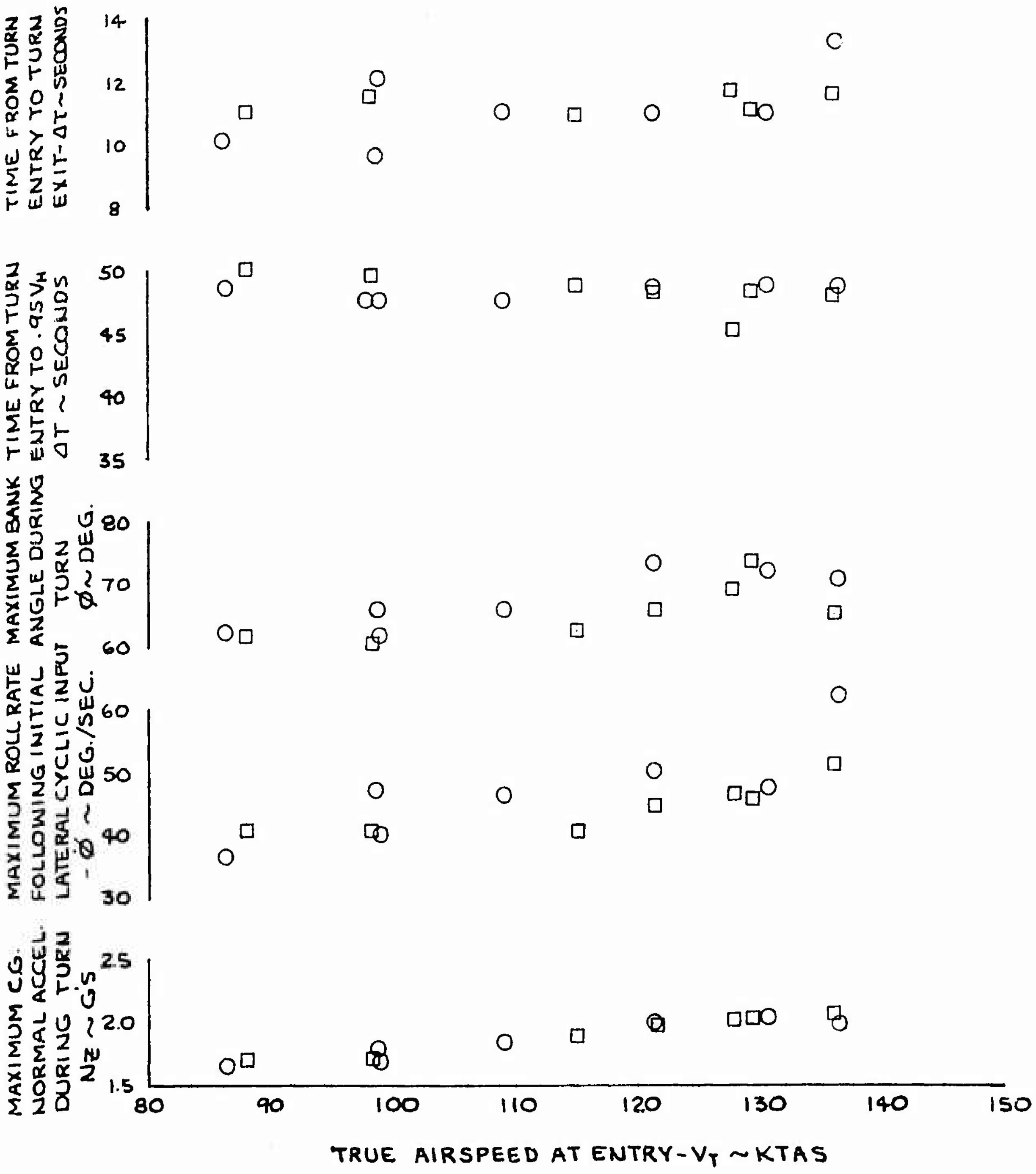


FIGURE 23
180 DEGREE TURNING PERFORMANCE
AH-1G T53-L-13

SYM	AVG. GRWT ~LB	AVG. ALT H_D~FT.	AVG. LONG C.G ~INCH	ROTOR SPEED ~RPM	ARMAMENT CONFIGURATION
□	9100	3000	196.2(MID)	324	HVY. HOG
○	8250	2400	195.6(MID)	324	HVY. HOG
△	8250	2400	193.1(MID)	324	CLEAN
◇	7250	3100	195.6(MID)	324	CLEAN

NOTES: 1. SHADED SYMBOLS DENOTE LEFT TURN
2. OPEN SYMBOLS DENOTE RIGHT TURN

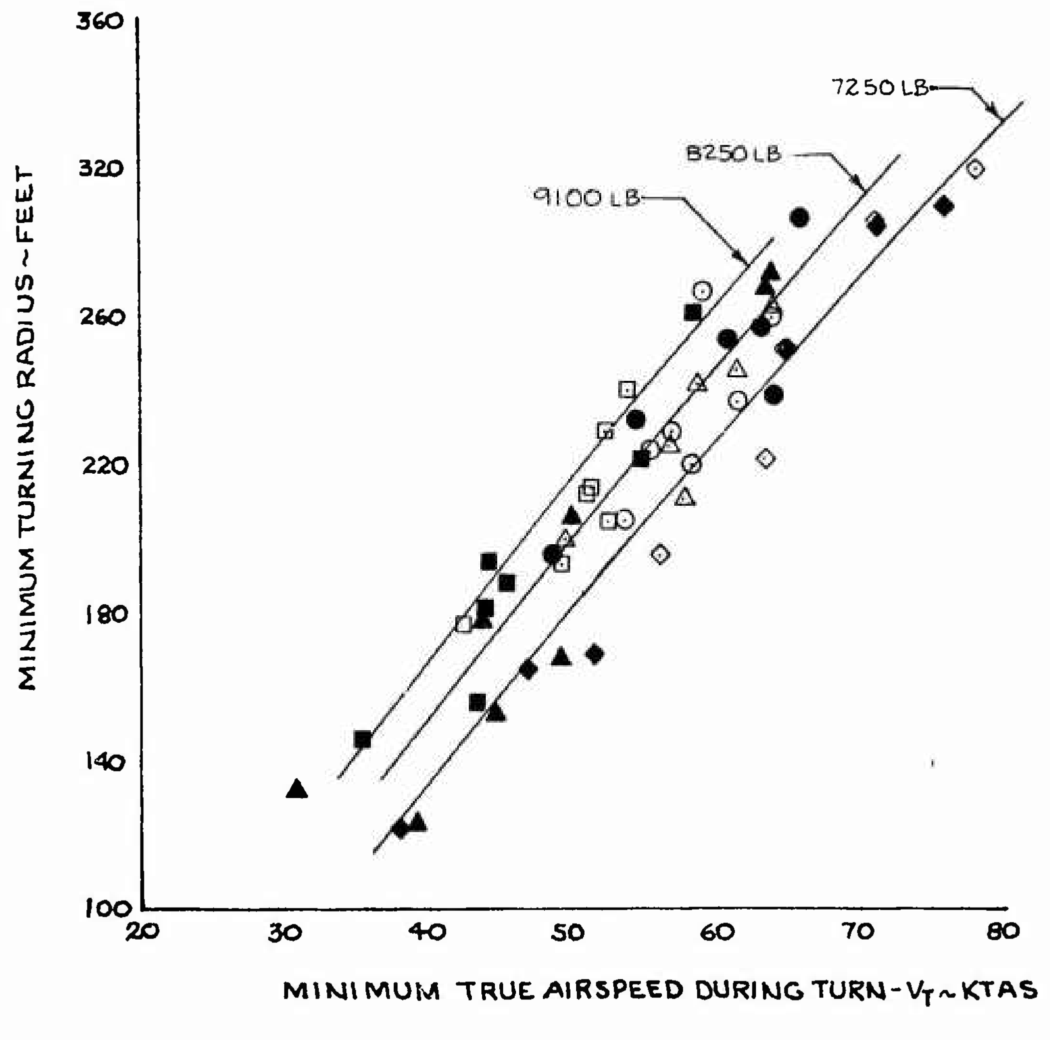


FIGURE 24
180° TURNING PERFORMANCE
AH-1G USA S/N 615247
CLEAN CONFIGURATION

GRWT ~LB	DENSITY ALTITUDE H_D ~ FEET	LONG C.G. POSITION ~ INCHES	ROTOR SPEED ~ RPM
8210	2600	193.0 (FWD)	322.5

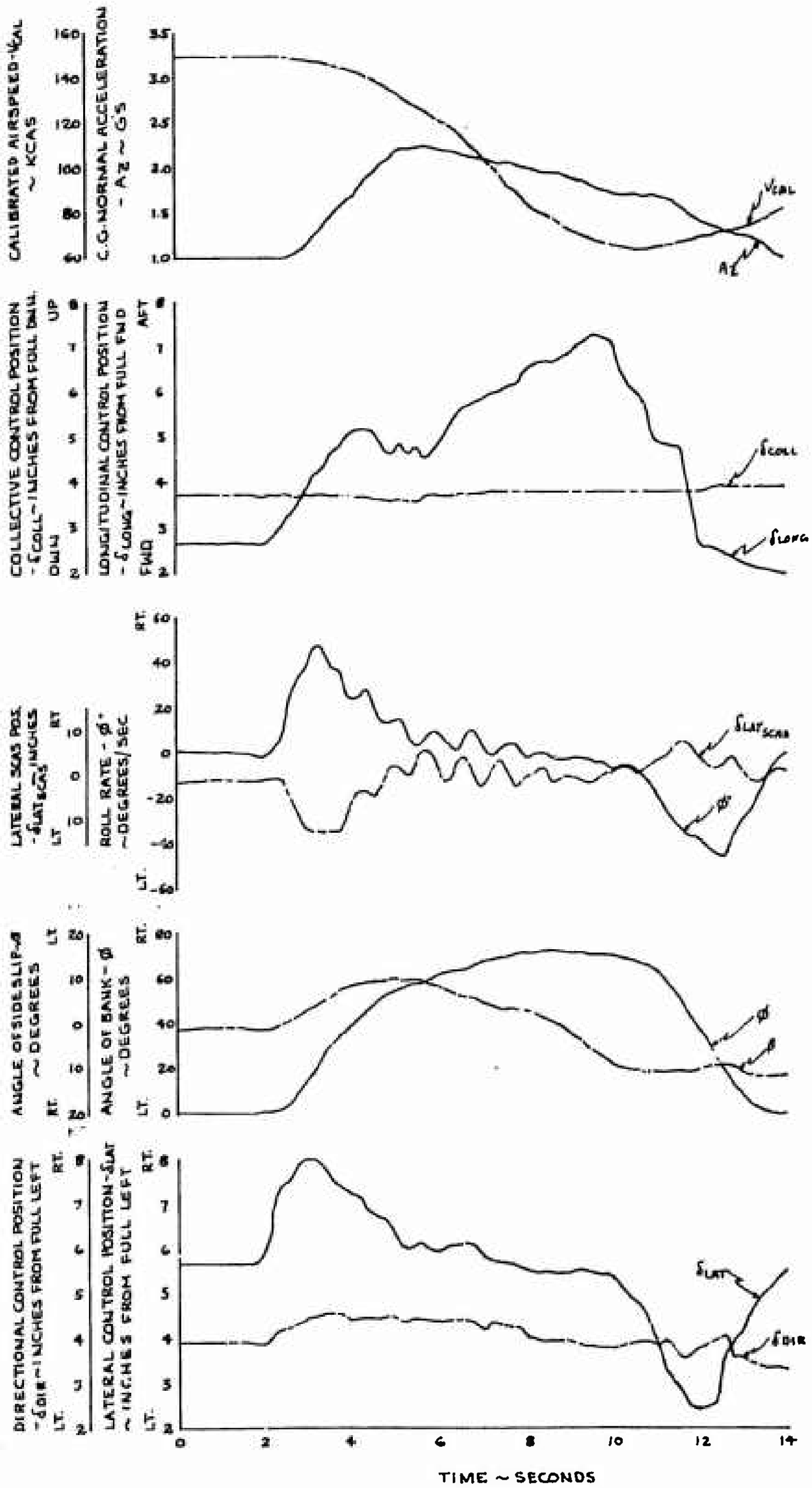


FIGURE 25
180° TURNING PERFORMANCE
AH-1G USA S/N 66-15247
CLEAN CONFIGURATION

GR WT ~LB	DENSITY ALTITUDE H_D ~FEET	LONG. C.G. LOCATION ~INCHES	ROTOR SPEED ~RPM
8190	2570	193.0 (FWD)	323.0

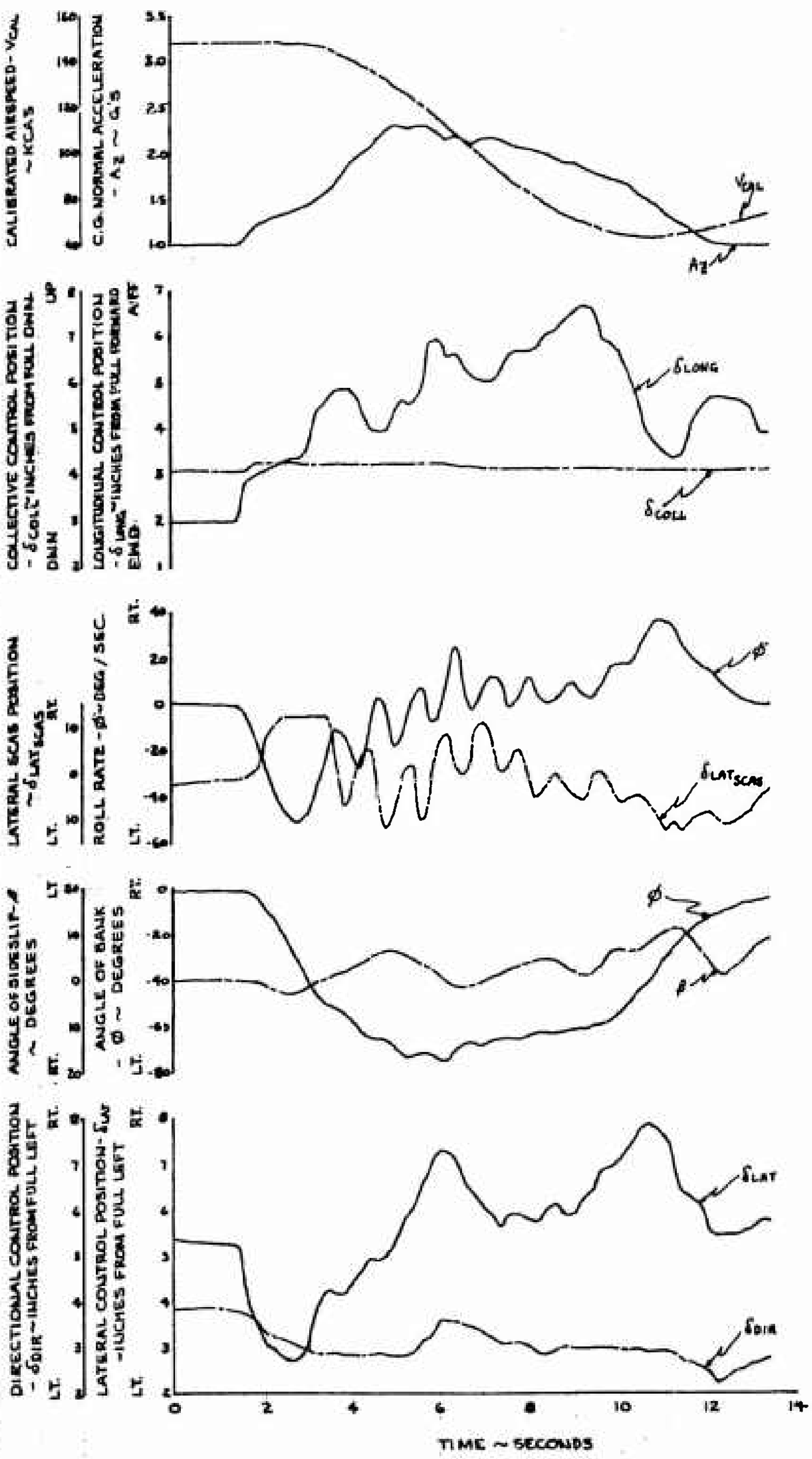


FIGURE 26

ALTITUDE LOSS DURING RECOVERY FROM DIVE

AH-1G T53-L-13

NOTES: 1. DATA DERIVED FROM FIGURES 27 THROUGH 31 APPENDIX V

2. DATA BASED ON THE FOLLOWING CONDITIONS:

a. ZERO ACCELERATION ALONG FLIGHT PATH AT START OF RECOVERY MANEUVER

b. RECOVERY MANEUVERS WERE CONDUCTED WITH LESS THAN 5 DEGREES OF ROLL ATTITUDE

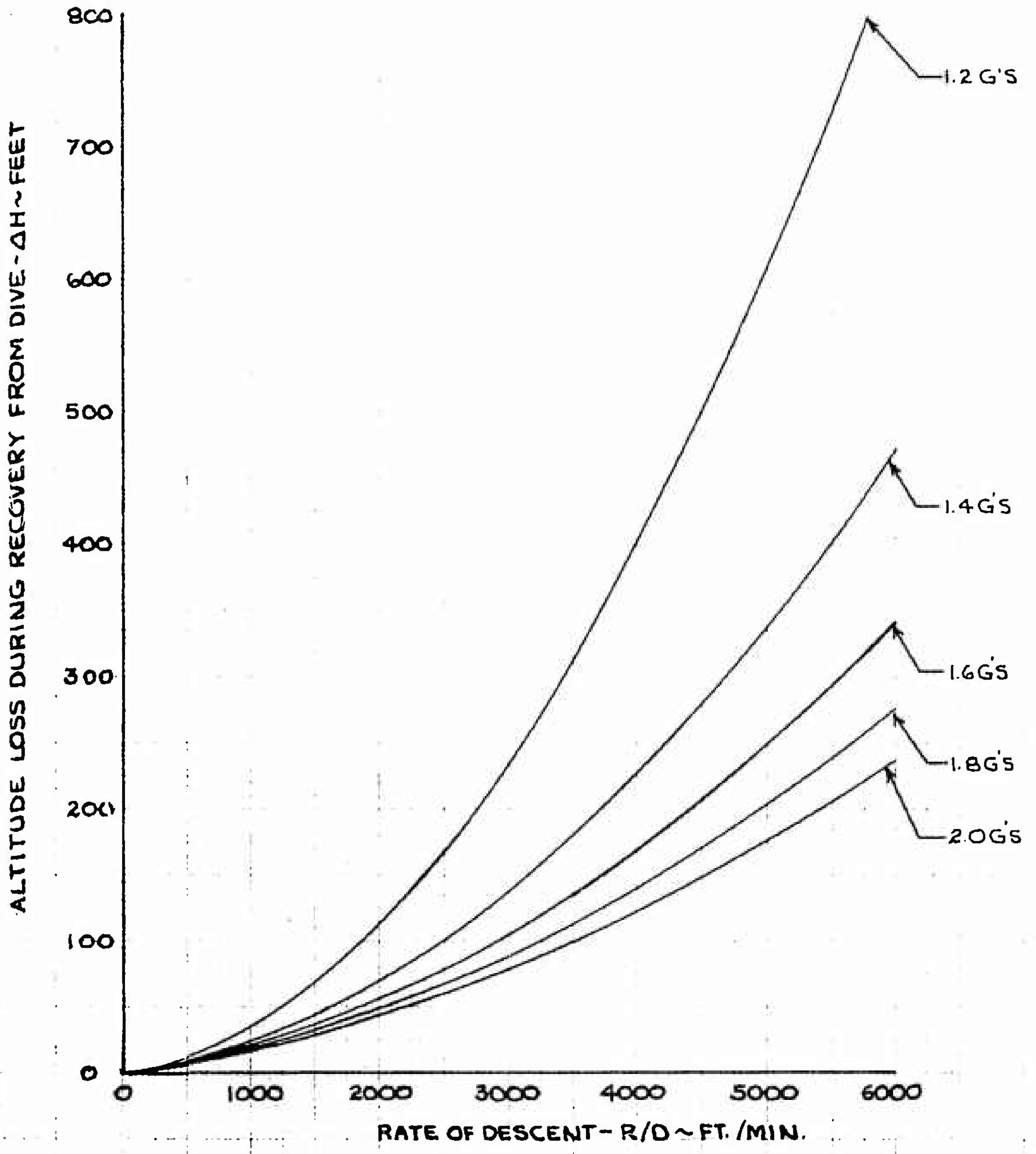


FIGURE 27

ALTITUDE LOSS DURING RECOVERY FROM DIVE

AH-1G T53-L-13

SYM.	CONFIG.	AVG. ALTITUDE H_D ~ FT.	AVG. GR.WT. ~LB.	AVG. LONG. C.G. ~INCH	ROTOR SPEED ~RPM	FLT. PATH ANGLE RANGE~DEG.
○	HVY. HOG	4300	8300	196.3(MID)	324	6.33 TO 13.55
△	HVY HOG	4700	8200	195.7(MID)	324	4.22 TO 14.32
◇	HVY. HOG	4700	9200	195.7(MID)	324	4.57 TO 12.95
□	CLEAN	5700	8200	194.6(MID)	324	5.82 TO 15.00

NOTES: 1. ALL RECOVERY MANEUVERS WERE CONDUCTED WITH LESS THAN FIVE DEGREES OF ROLL ATTITUDE

2. ALL RECOVERY MANEUVERS WERE INITIATED WITH ZERO ACCELERATION ALONG THE FLIGHT PATH

3. DATA IS VALID FOR FLIGHT PATH ANGLES LESS THAN 15 DEGREES

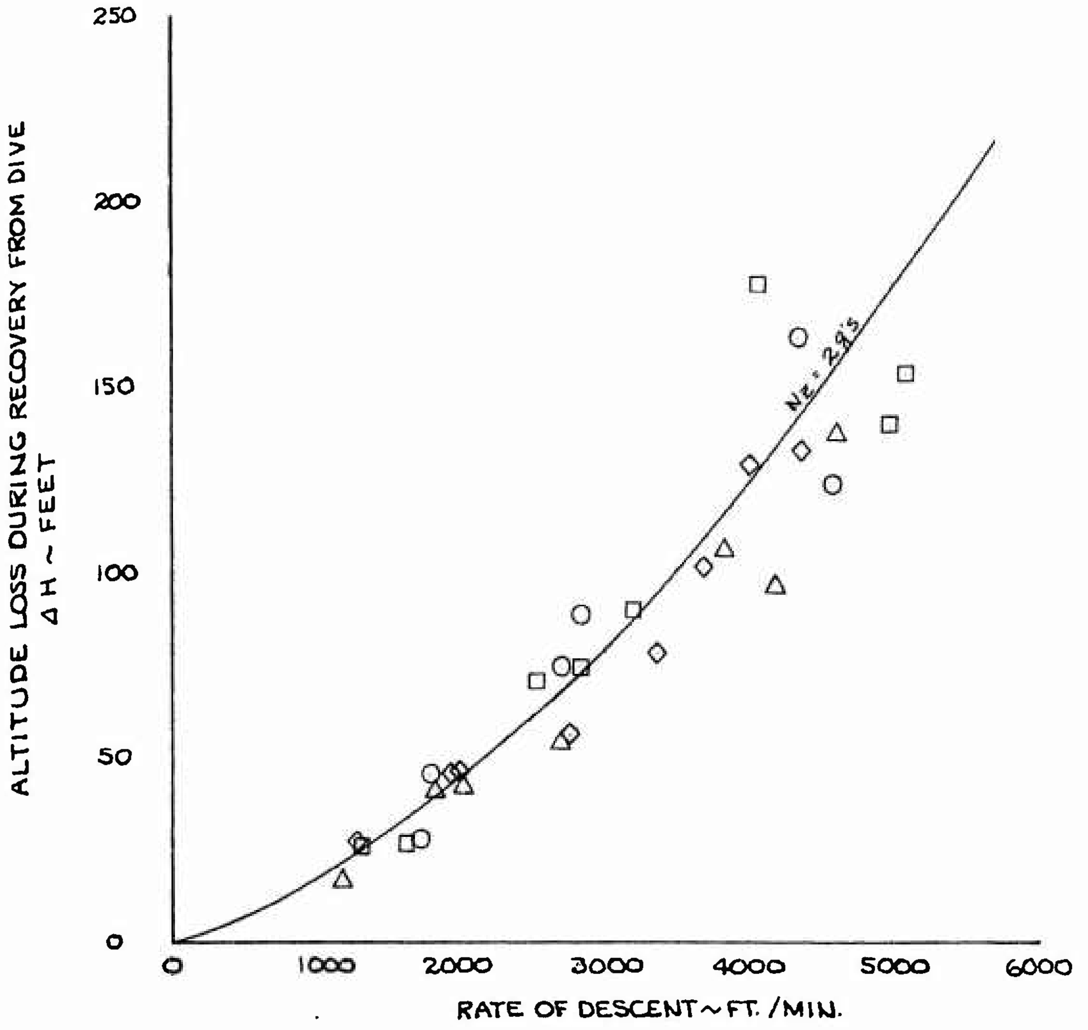


FIGURE 28
ALTITUDE LOSS DURING RECOVERY FROM DIVE
AH-1G T53-L-13

CLEAN CONFIGURATION

SYM.	AVG. ALTITUDE H_D ~ FEET	AVG. GRWT ~ LB.	AVG. LONG C.G. ~ INCH	ROTOR SPEED ~ RPM	RATE OF DESCENT ~ FT./MIN	TRUE AIRSPEED AT RECOVERY ~ KTAS
○	5700	8500	194.9 (MID)	324	1800	159.5
△	5700	8300	194.7 (MID)	324	2910	179.0
□	5700	8100	194.4 (MID)	324	4770	193.0

NOTE: DATA BASED ON THE FOLLOWING CONDITIONS:
a. ZERO ACCELERATION ALONG FLIGHT PATH AT START OF RECOVERY MANEUVER
b. RECOVERY MANEUVERS WERE CONDUCTED WITH LESS THAN 5 DEGREES OF ROLL ATTITUDE

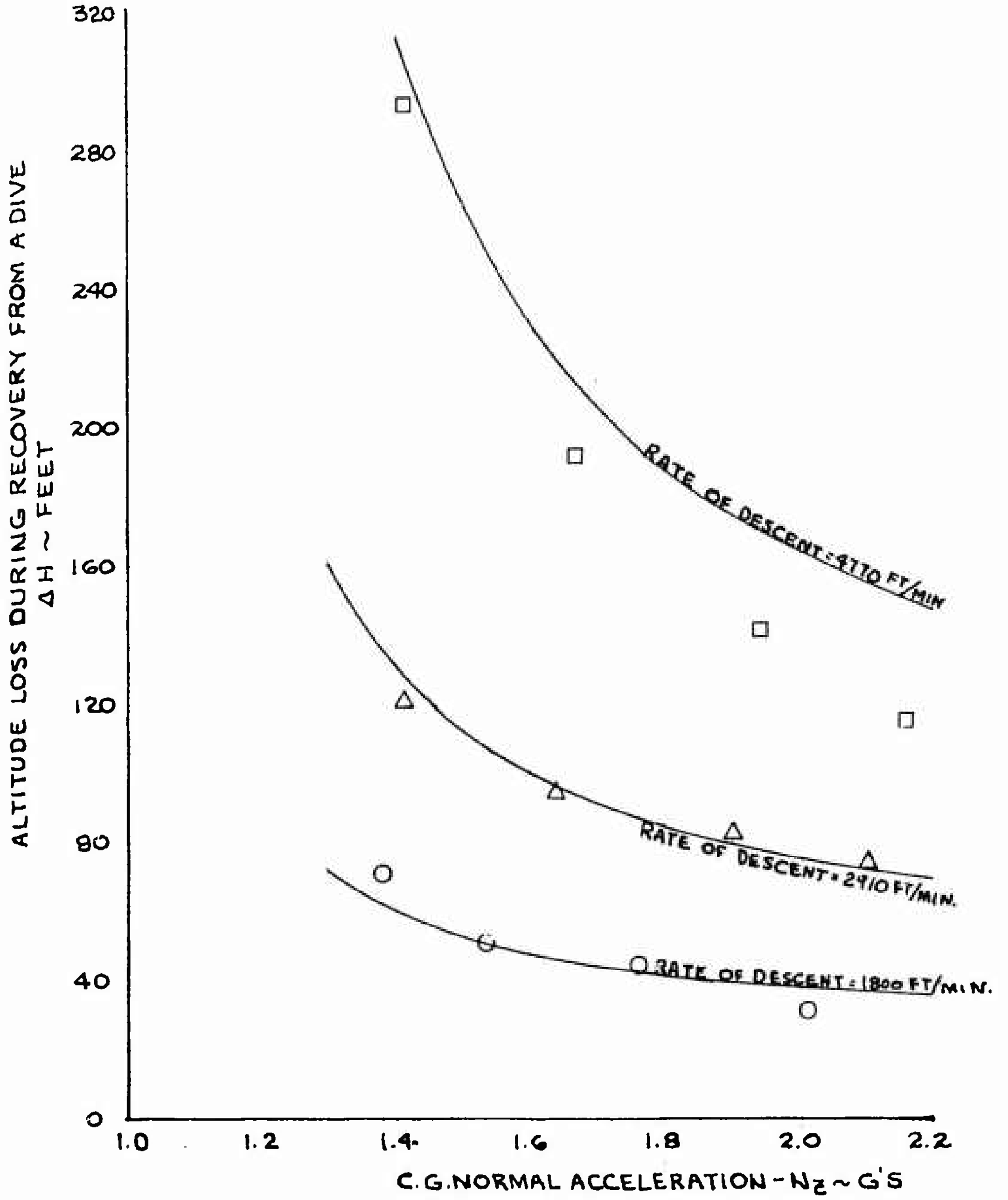


FIGURE 29

ALTITUDE LOSS DURING RECOVERY FROM DIVE

AH-1G T53-L-13

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SYM.	AVG. ALTITUDE H_D ~FEET	AVG. GR. WT. ~ LB.	AVG. LONG. C.G. ~INCH	ROTOR SPEED ~ RPM	RATE OF DESCENT ~FT./MIN.	TRUE AIRSPEED AT RECOVERY~KTAS
○	4700	8300	195.8(MID)	324	2050	153.5
△	4700	8000	195.6(MID)	324	3580	173.0
□	4700	7800	195.4(MID)	324	4850	187.5

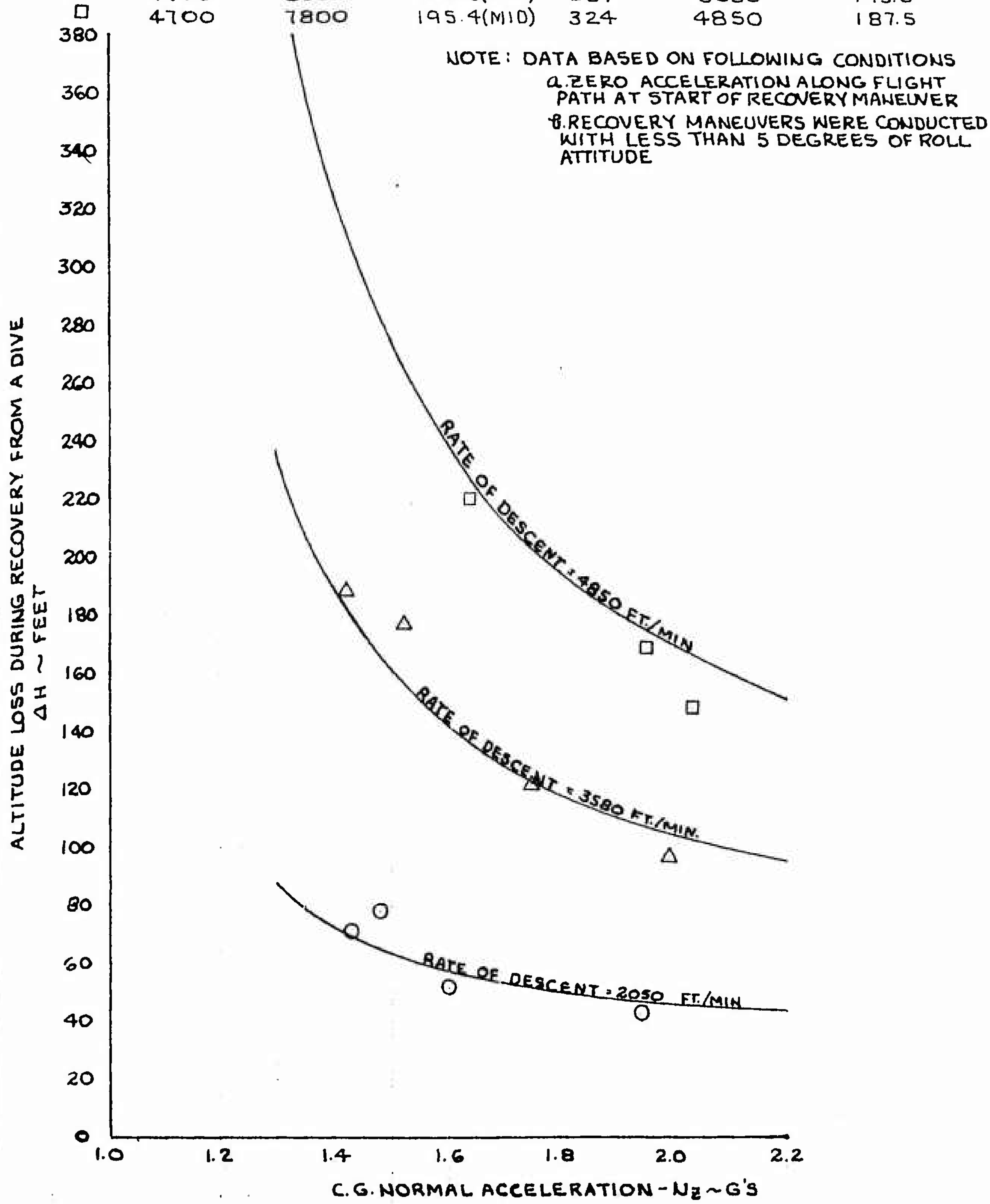


FIGURE 30

ALTITUDE LOSS DURING RECOVERY FROM DIVE

AH-1G T53-L-13

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SYM.	AVG. ALTITUDE H_D ~FEET	AVG. GRWT ~LB.	AVG. LONG. CG ~INCH	ROTOR SPEED ~RPM	RATE OF DESCENT ~FT./MIN	TRUE AIRSPEED AT RECOVERY~KTAS
O	4300	8600	196.6(MID)	324	1720	156
Δ	4300	8300	196.4(MID)	324	2750	174
□	4300	8100	196.2(MID)	324	4170	193.5

NOTE: DATA BASED ON THE FOLLOWING CONDITIONS
a. ZERO ACCELERATION ALONG FLIGHT PATH AT START OF RECOVERY MANEUVER
b. RECOVERY MANEUVERS WERE CONDUCTED WITH LESS THAN 5 DEGREES OF ROLL ATTITUDE

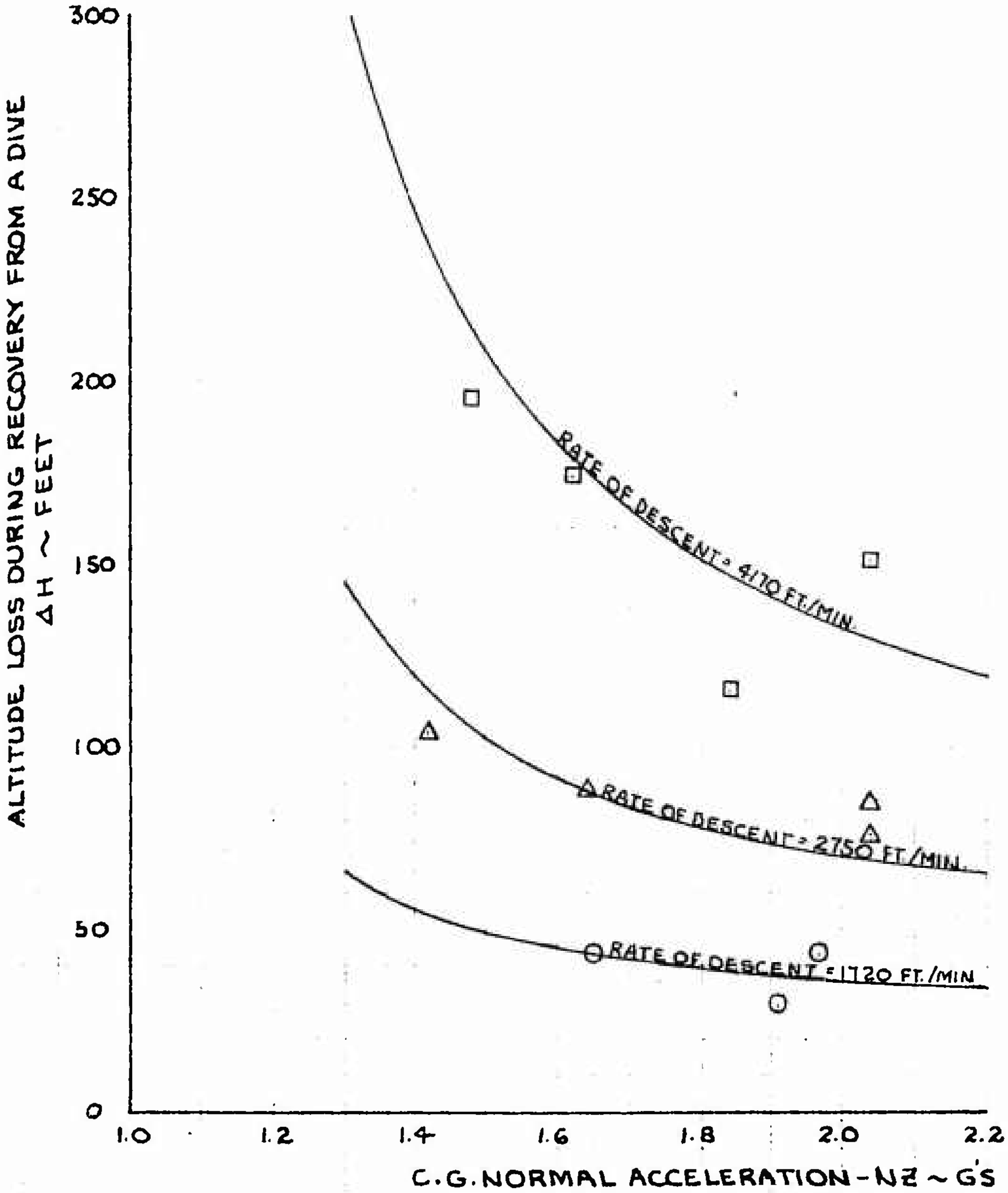


FIGURE 31

ALTITUDE LOSS DURING RECOVERY FROM DIVE

AH-1G T53-L-13

HVY. HOG CONFIGURATION WITH ROCKET POD FAIRINGS REMOVED

SYM.	AVG. ALTITUDE H_D ~ FEET	AVG. GRWT ~LB.	AVG. LONG. C.G. ~INCH	ROTOR SPEED ~RPM	RATE OF DESCENT ~FT./MIN.	TRUE AIRSPEED AT RECOVERY~KTAS
○	4700	9100	195.8(MID)	324	2100	154.0
△	4700	8900	195.6(MID)	324	3480	178.5
□	4700	8700	195.4(MID)	324	4220	191.0

NOTE : DATA BASED ON THE FOLLOWING CONDITIONS :

a. ZERO ACCELERATION ALONG FLIGHT PATH AT START OF RECOVERY MANEUVER

b. RECOVERY MANEUVERS WERE CONDUCTED WITH LESS THAN 5 DEGREES OF ROLL ATTITUDE

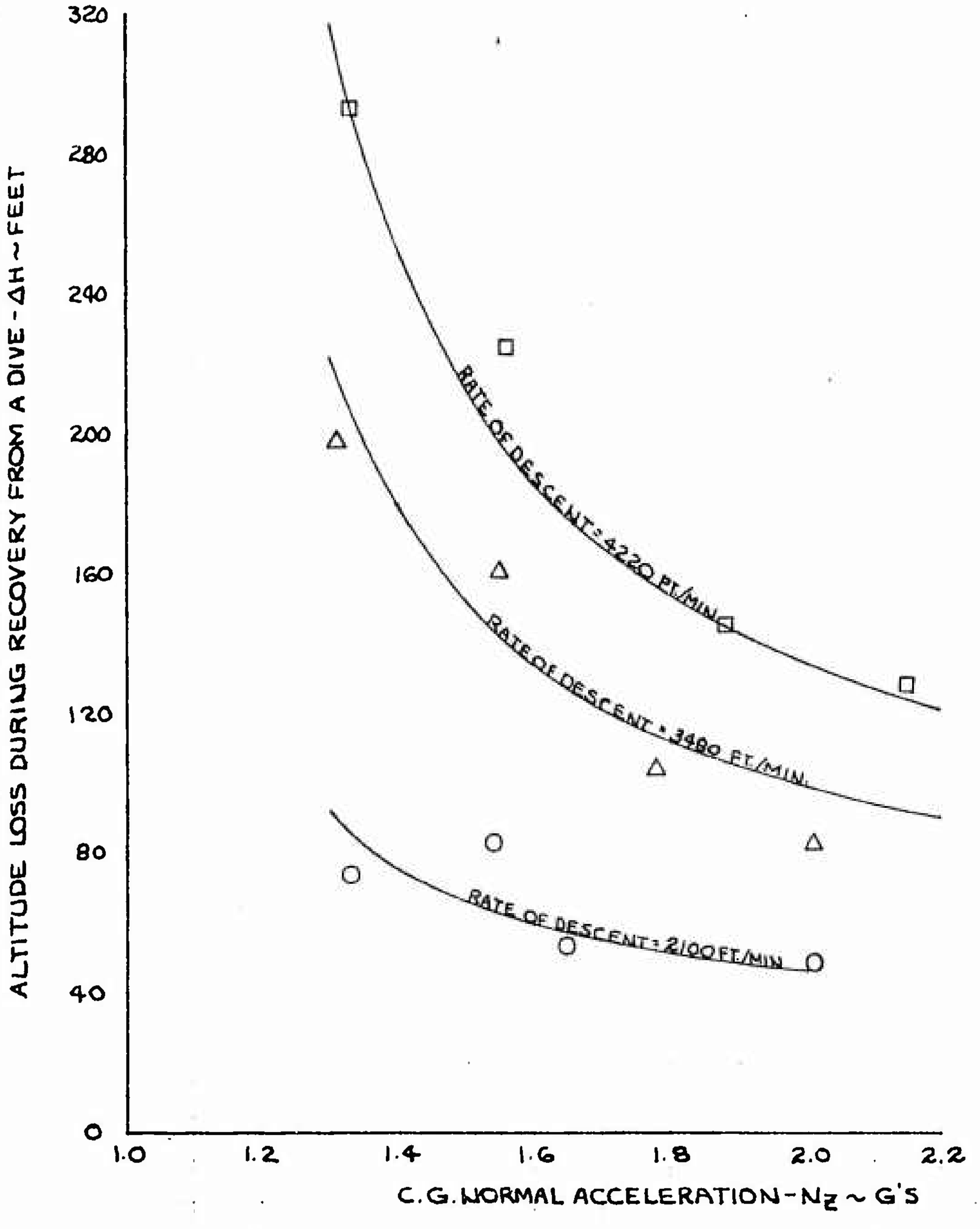


FIGURE 32
ALTITUDE LOSS DURING RECOVERY FROM DIVE
AH-1G T53-L-13

NOTES: 1. DATA BASED ON FOLLOWING CONDITIONS
a. ZERO ACCELERATION ALONG FLIGHT PATH AT START OF RECOVERY MANEUVER
b. RECOVERY MANEUVERS WERE CONDUCTED WITH LESS THAN 5 DEGREES OF ROLL ATTITUDE
2. γ DENOTES FLIGHT PATH ANGLE

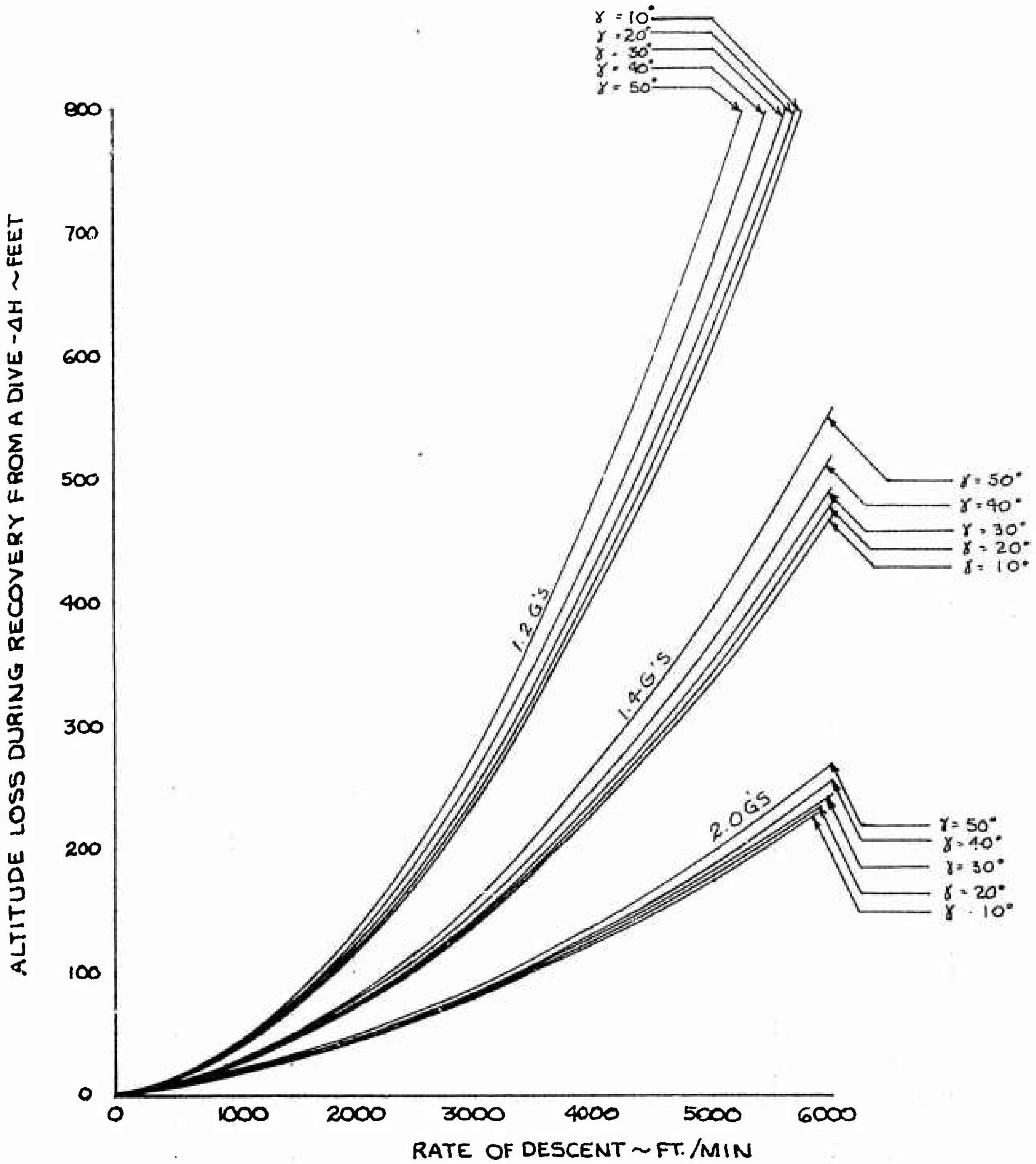


FIGURE 33

DIVING FLIGHT AND PULL UP MANEU

AH-1G USA S/N 615247

HVY. HOG CONFIGURATION WITH ROCKET POD FAI

GRWT = 7710 LB. LONG. C.G. POSITION = 195.3 I

NOTES: 1. COLLECTIVE CONTROL POSITION REMAINED FIXED THROUGH

2. FAIRED DATA WAS OBTAINED FROM OSCILLOGRAPH

3. SYMBOLIZED DATA WAS OBTAINED FROM PHOTO PANEL

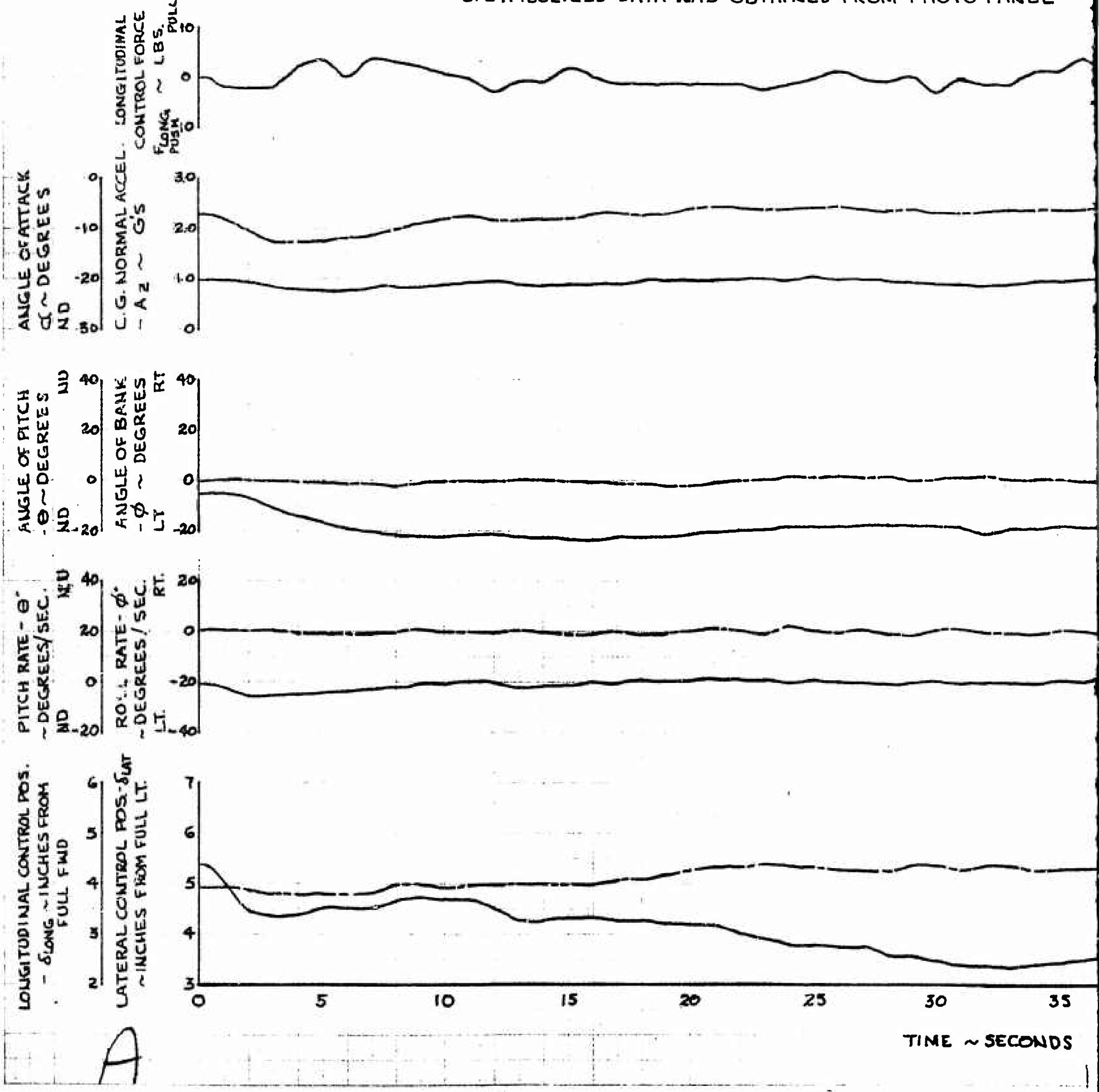


FIGURE 33

ɜHT AND PULL UP MANEUVER

ᴧ-1G USA S/N 615247

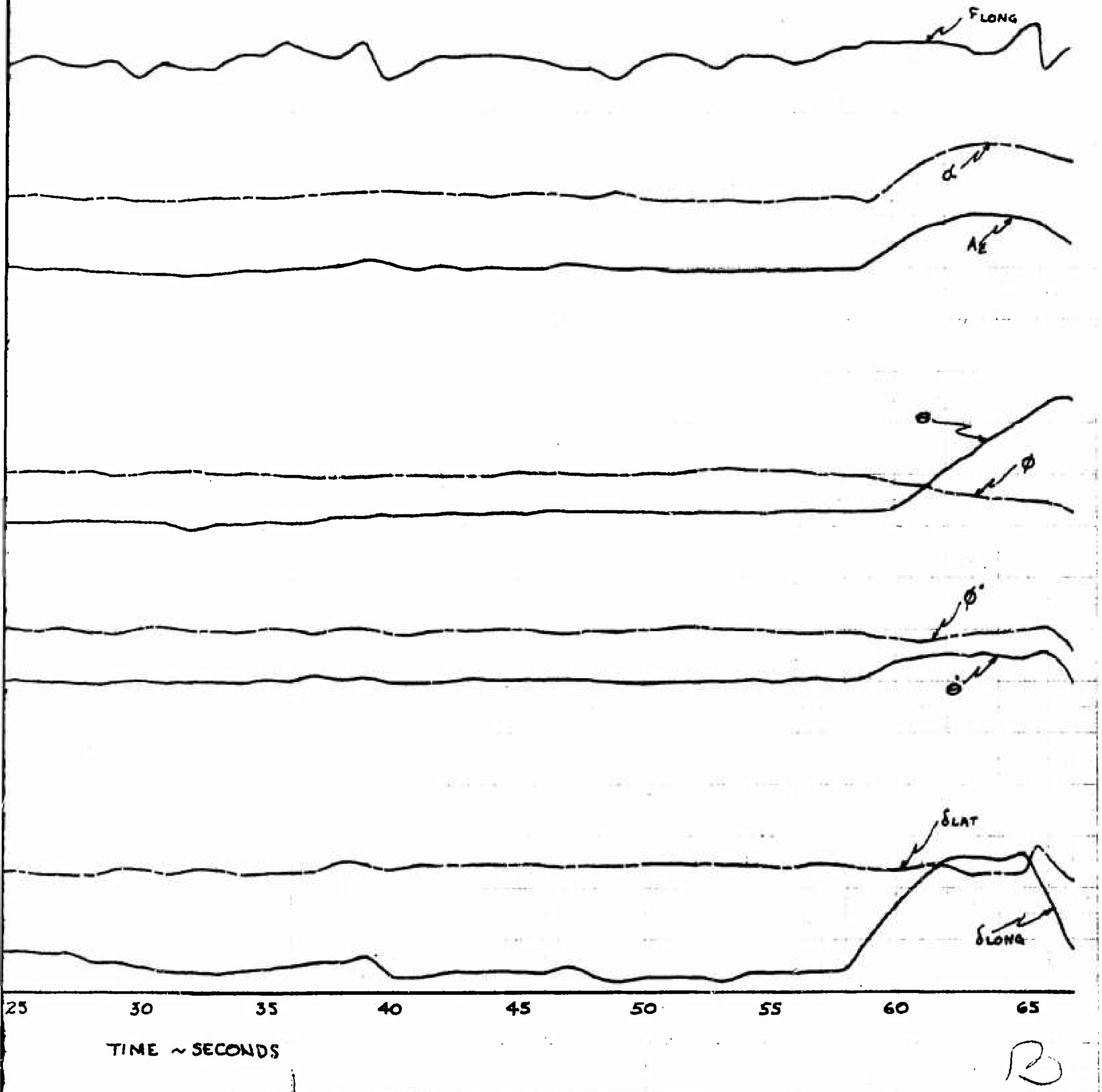
ᴦION WITH ROCKET POD FAIRINGS REMOVED

ᴊG. C.G. POSITION = 195.3 INCHES (MID)

ʐEMAINED FIXED THROUGH OUT ENTIRE MANEUVER

ᴧ OSCILLOGRAPH

ᴇD FROM PHOTO PANEL



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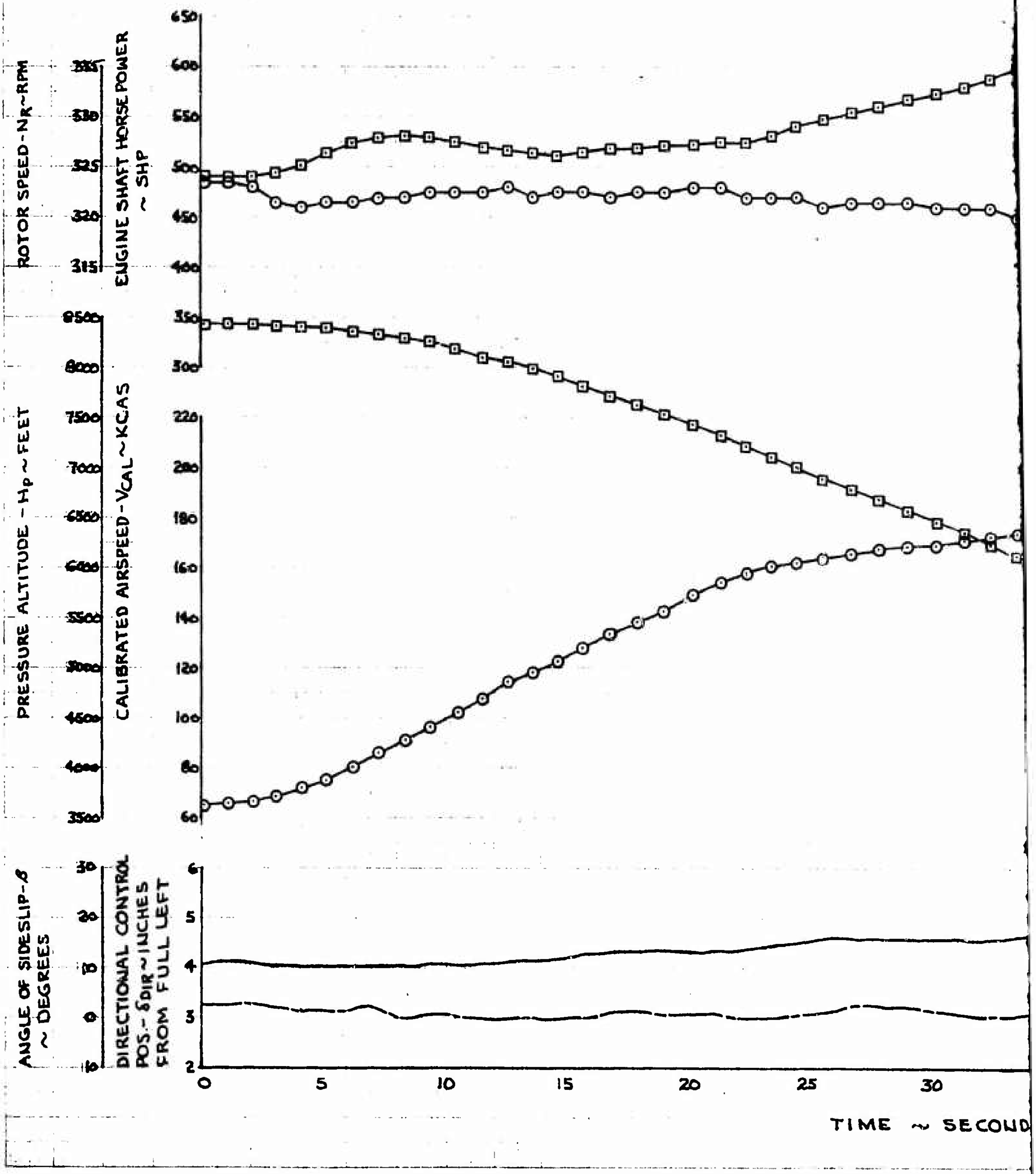

ROTOR SPEED - N_R ~ RPM
ENGINE SHAFT HORSE POWER ~ SHP
PRESSURE ALTITUDE - H_P ~ FEET
CALIBRATED AIRSPEED - V_{CAL} ~ KCAS
ANGLE OF SIDESLIP - β ~ DEGREES
DIRECTIONAL CONTROL POS. - δ_{DIR} ~ INCHES FROM FULL LEFT
TIME ~ SECONDS

FIGURE 33 (CONTINUED)

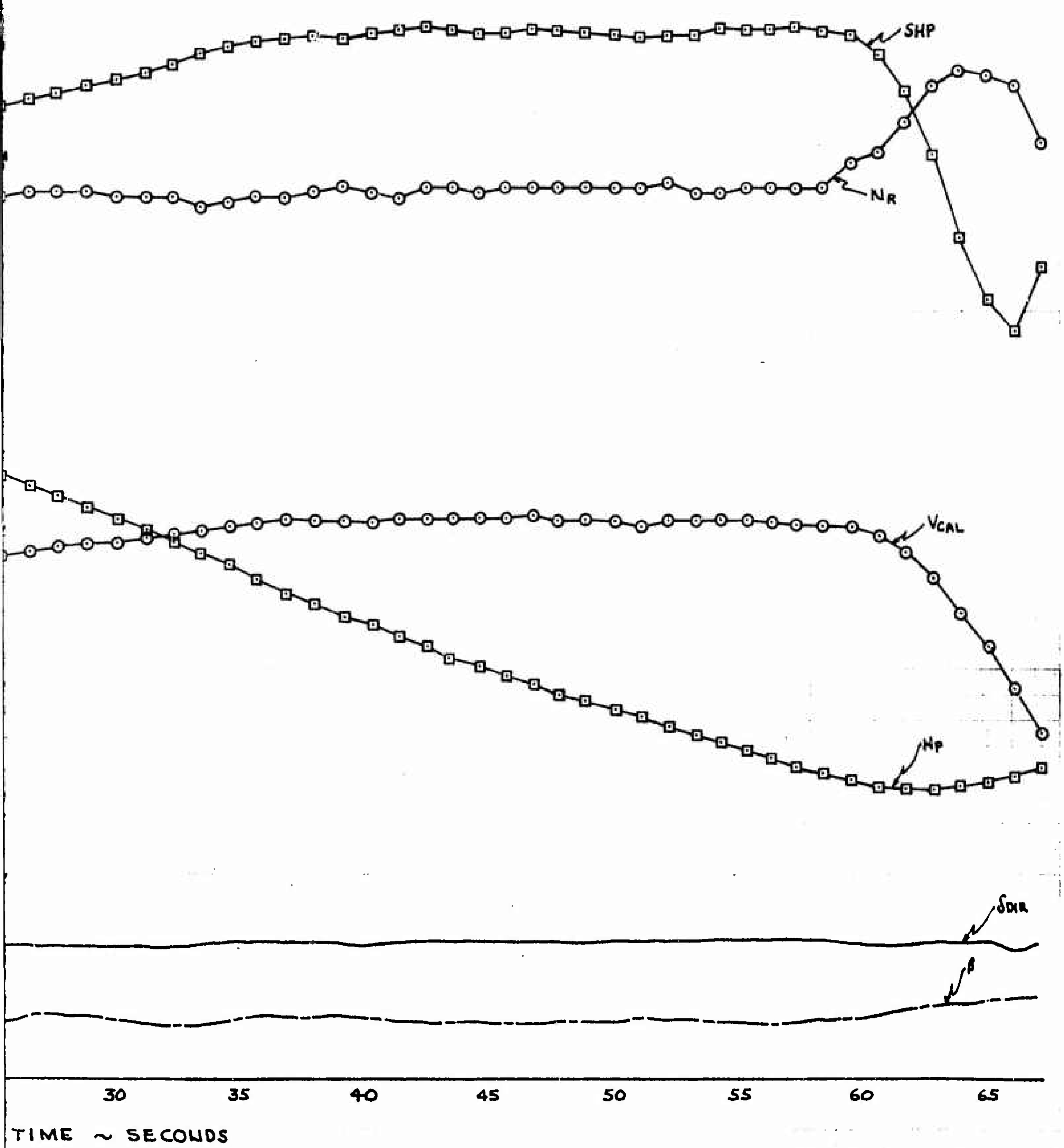


FIGURE 34

DIVING FLIGHT AND PULL U

AH-1G USA S/N 61 52

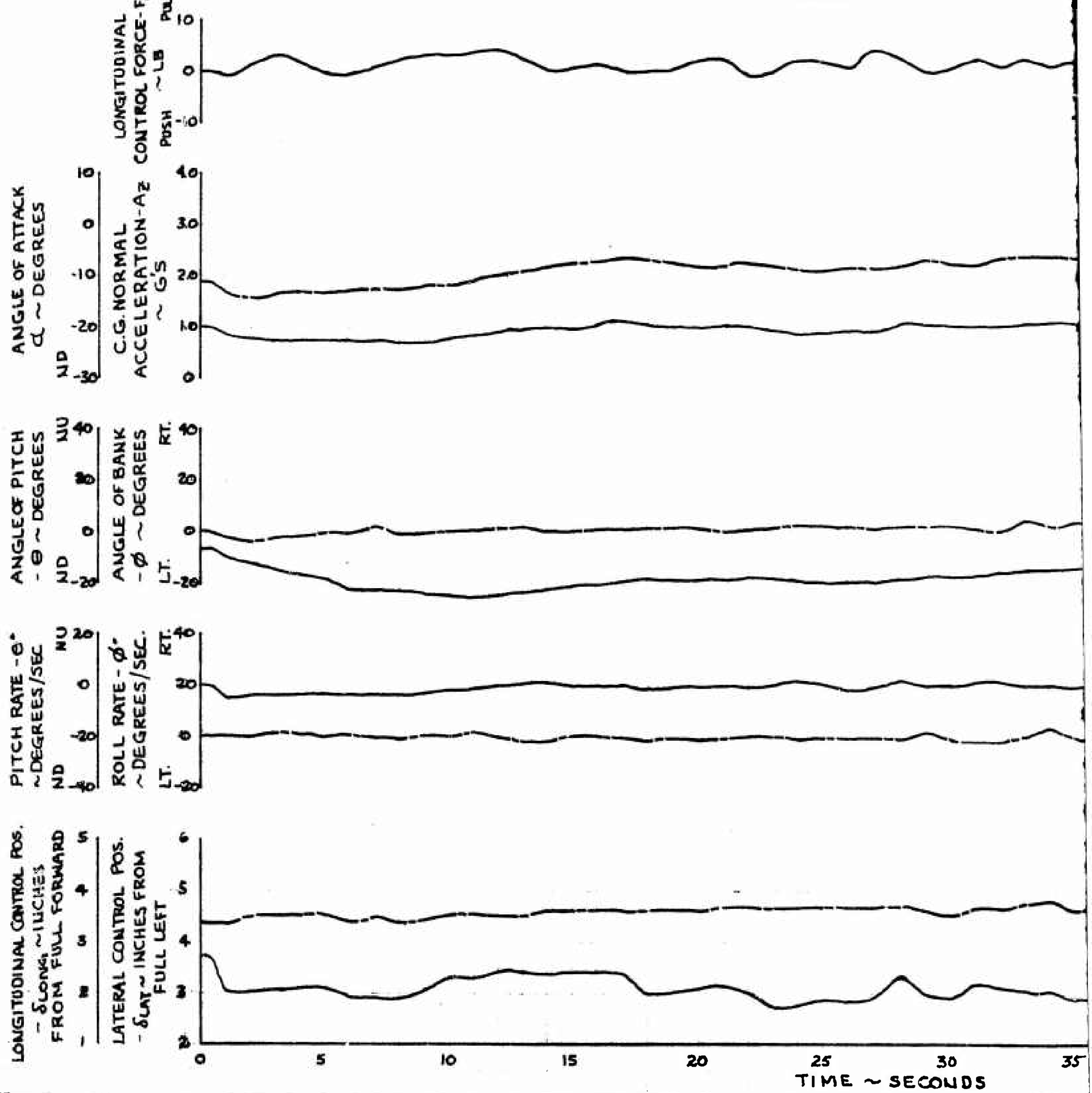
HVY. HOG CONFIGURATION WITH ROCKET

GRWT = 7995 LB. LONG. C.G. POSITION = 195

NOTES: 1. COLLECTIVE CONTROL POSITION REMAINED FIXED THROU

2. FAIRED DATA OBTAINED FROM OSCILLOGRAPH

3. SYMBOLIZED DATA WAS OBTAINED FROM PHOTO PANEL



A

URE 34

ND PULL UP MANEUVER

SA S/N 61 5247

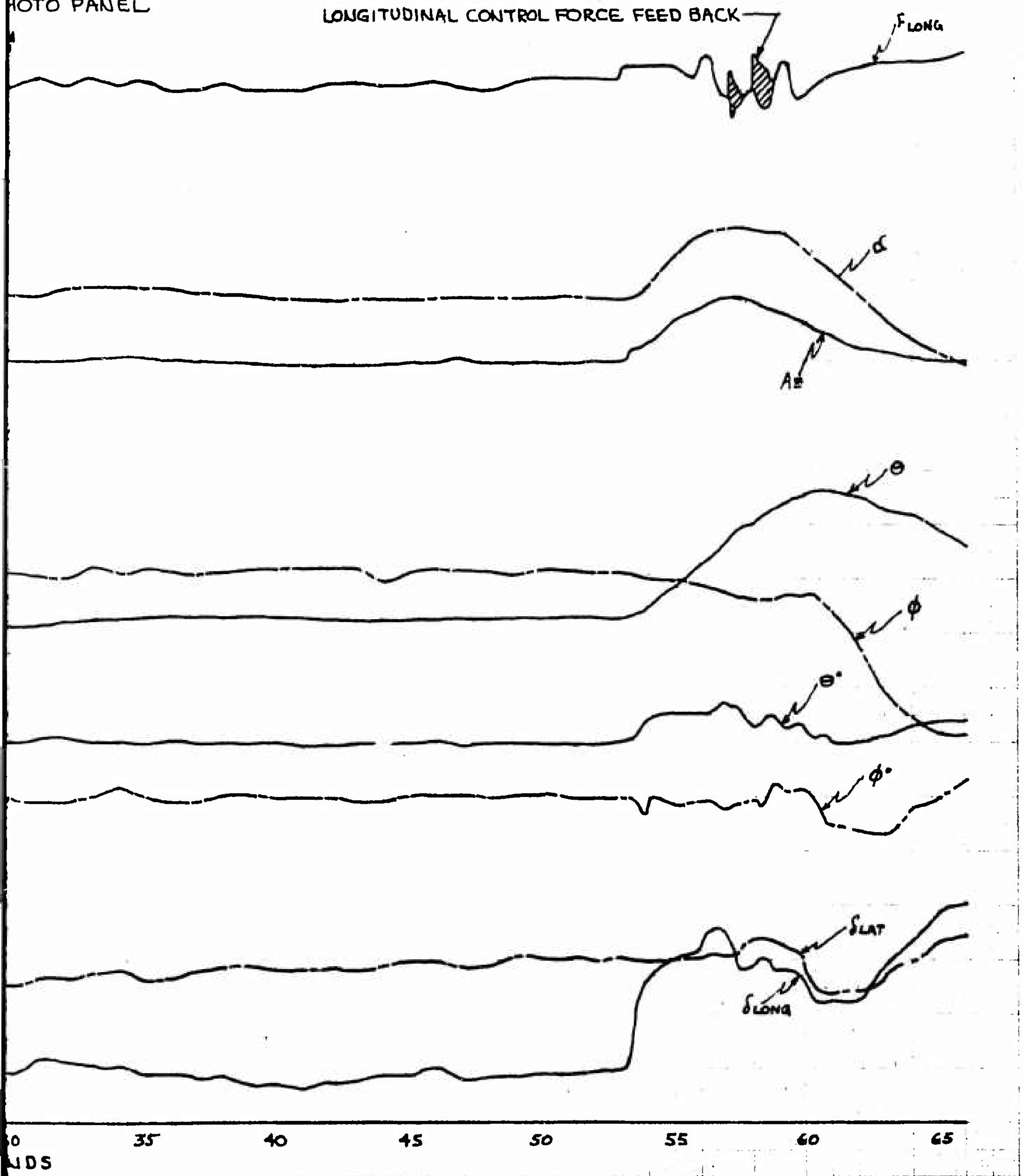
TH ROCKET POD FAIRINGS REMOVED

SITION = 195.6 INCHES (MID)

IXED THROUGHOUT ENTIRE MANEUVER

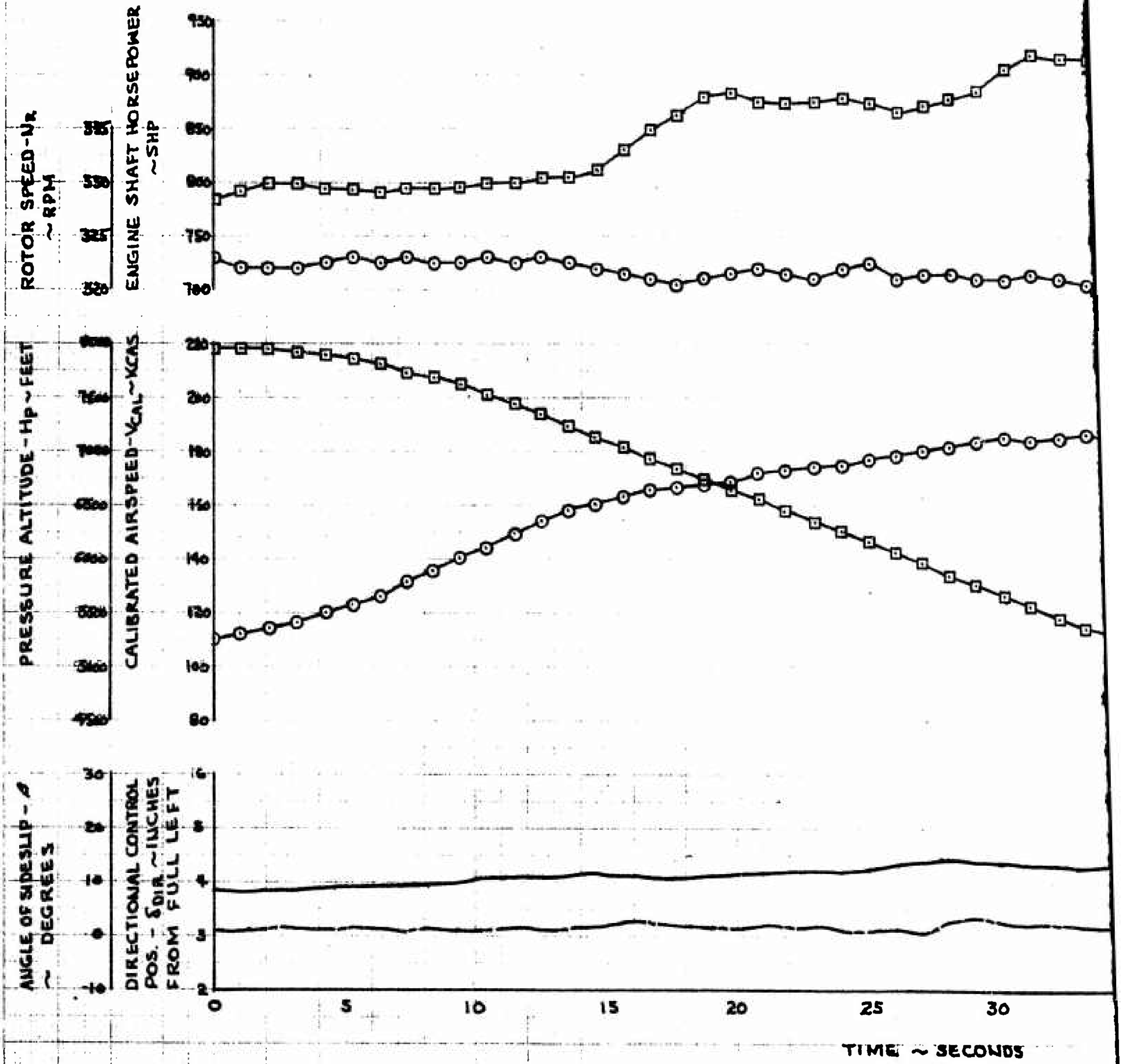
H

HOTO PANEL



B

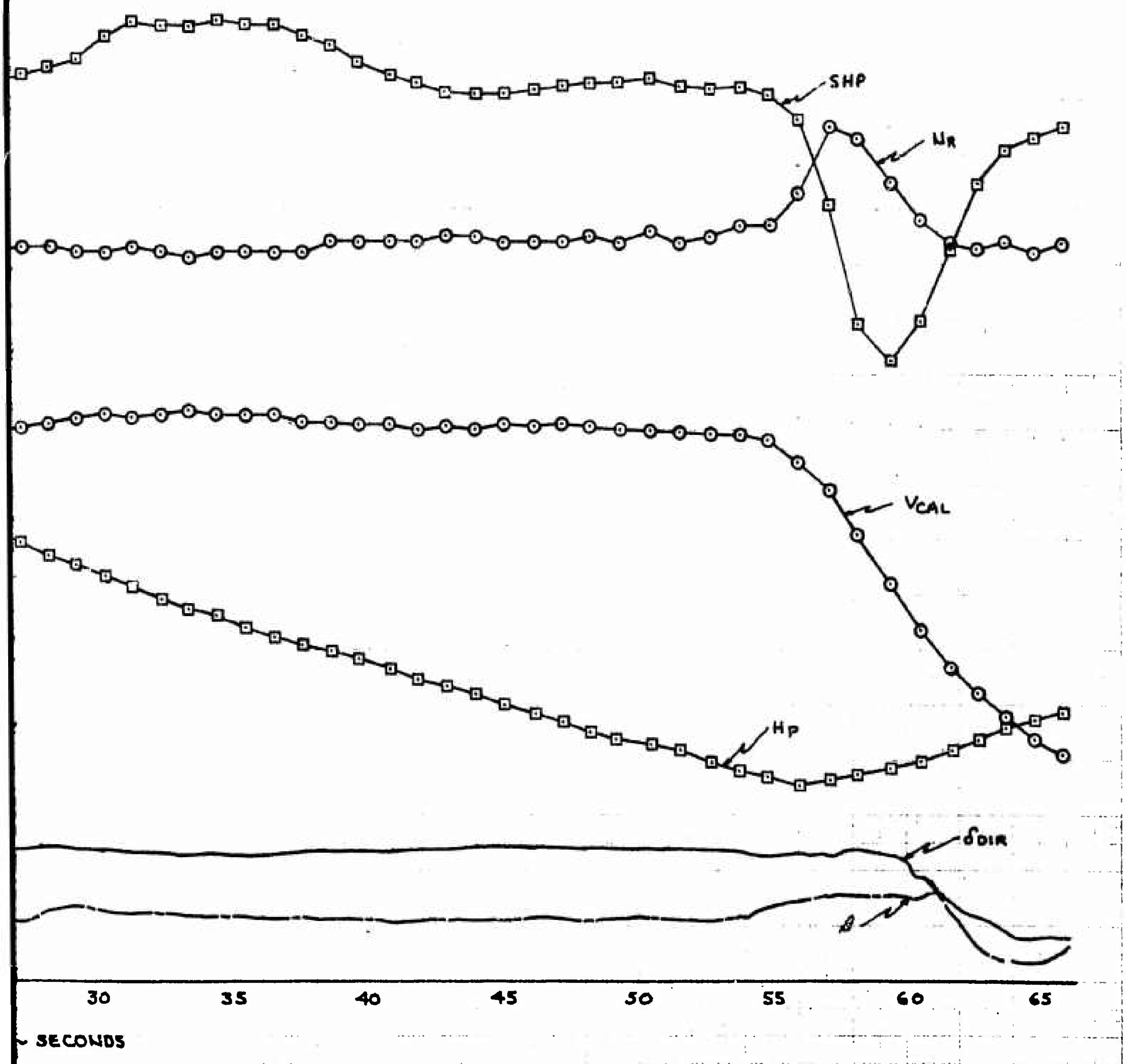
FIGURE 34 (



ROTOR SPEED-N_R ~RPM
ENGINE SHAFT HORSEPOWER ~SHP
PRESSURE ALTITUDE-H_P~FEET
CALIBRATED AIRSPEED-V_{CAL}~KCAS
ANGLE OF SIDESLIP - β ~ DEGREES
DIRECTIONAL CONTROL POS. - $\delta_{DIR.}$ ~INCHES FROM FULL LEFT
0
5
10
15
20
25
30
TIME ~ SECONDS

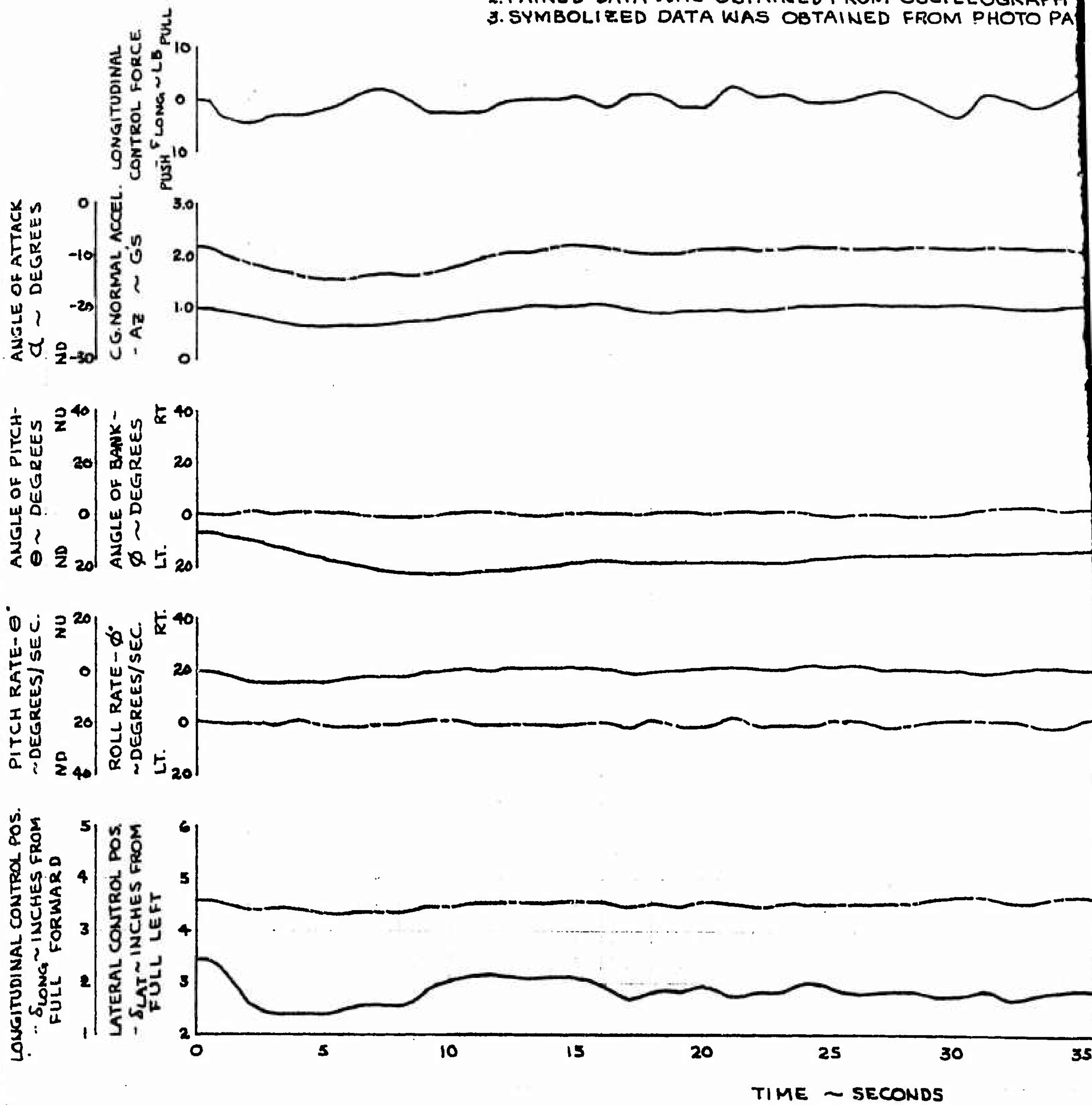
A

FIGURE 34 (CONTINUED)



B

FIGURE 35
DIVING FLIGHT AND PULL UP M
AH-1G USA S/N 615247
HVY. HOG CONFIGURATION WITH ROCKET
GRWT = 8415 LB LONG. C.G. POSITION = 195
NOTES: 1. COLLECTIVE CONTROL POSITION REMAINED FIXE
2. FAIRED DATA WAS OBTAINED FROM OSCILLOGRAPH
3. SYMBOLIZED DATA WAS OBTAINED FROM PHOTO PA



IGURE 35

AND PULL UP MANEUVER

USA S/N 615247

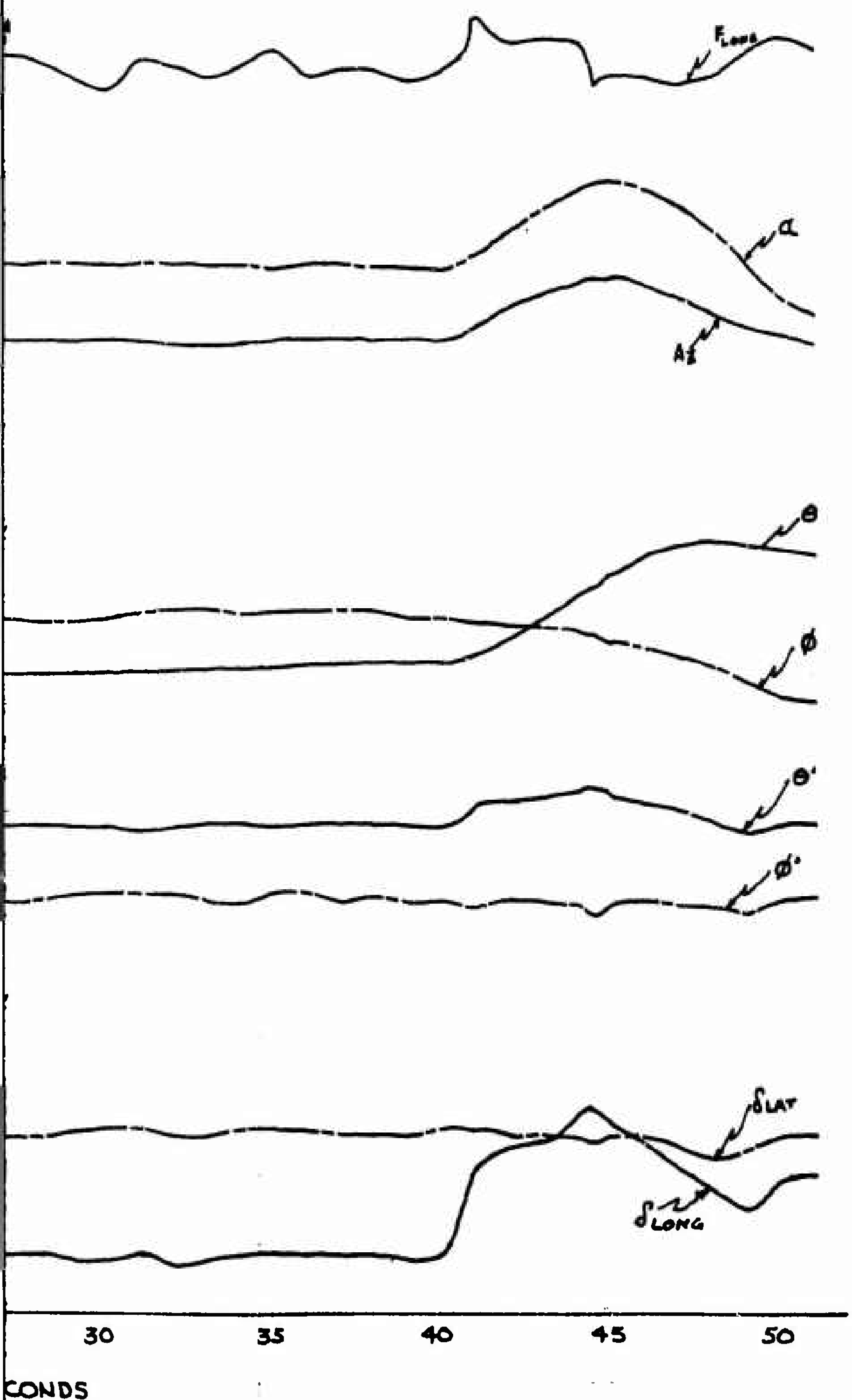
N WITH ROCKET POD FAIRINGS REMOVED

POSITION = 195.9 INCHES (MID)

REMAINED FIXED THROUGHOUT ENTIRE MANEUVER

OSCILLOGRAPH

FROM PHOTO PANEL



B

FIGURE 35 (CON

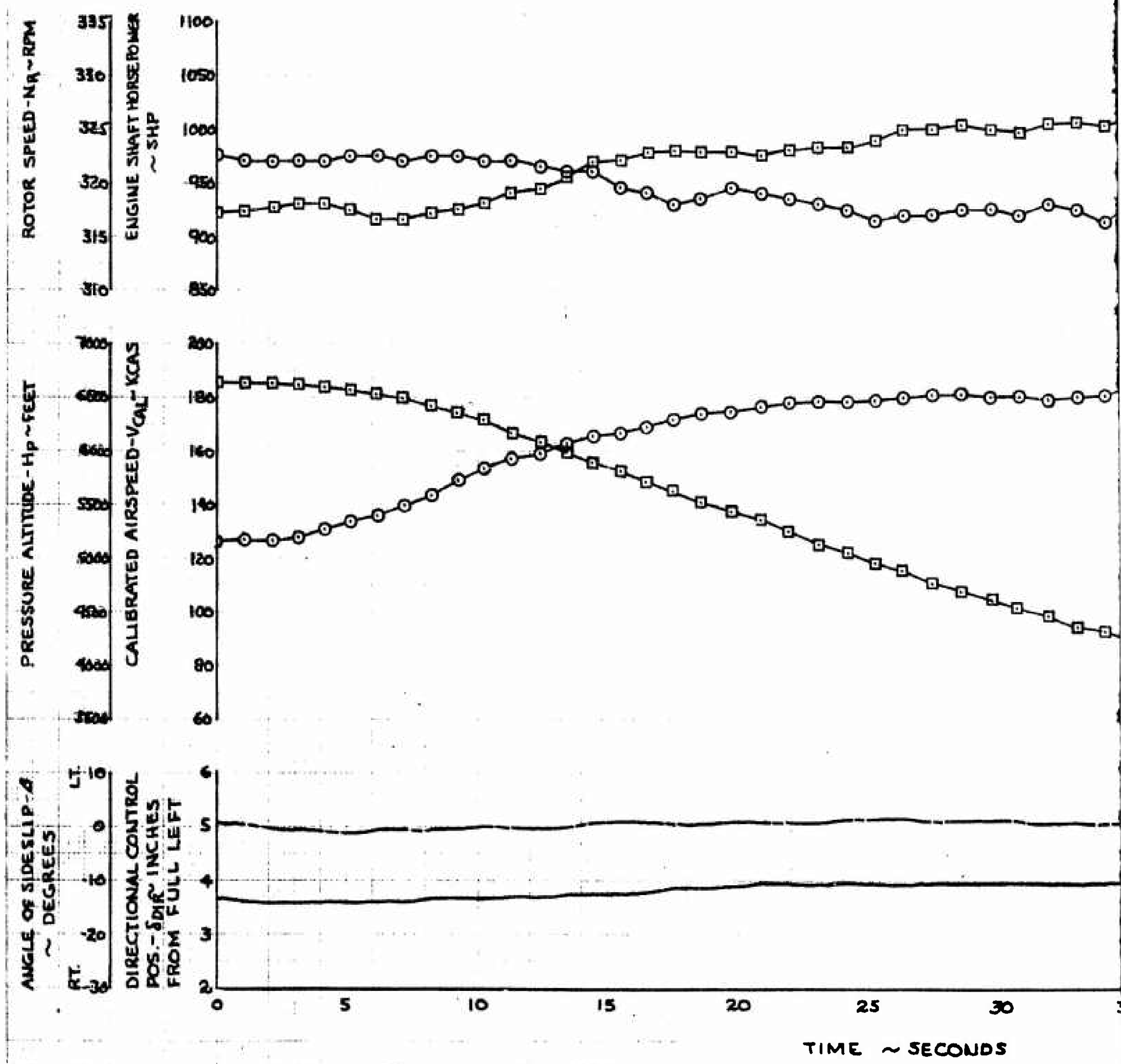
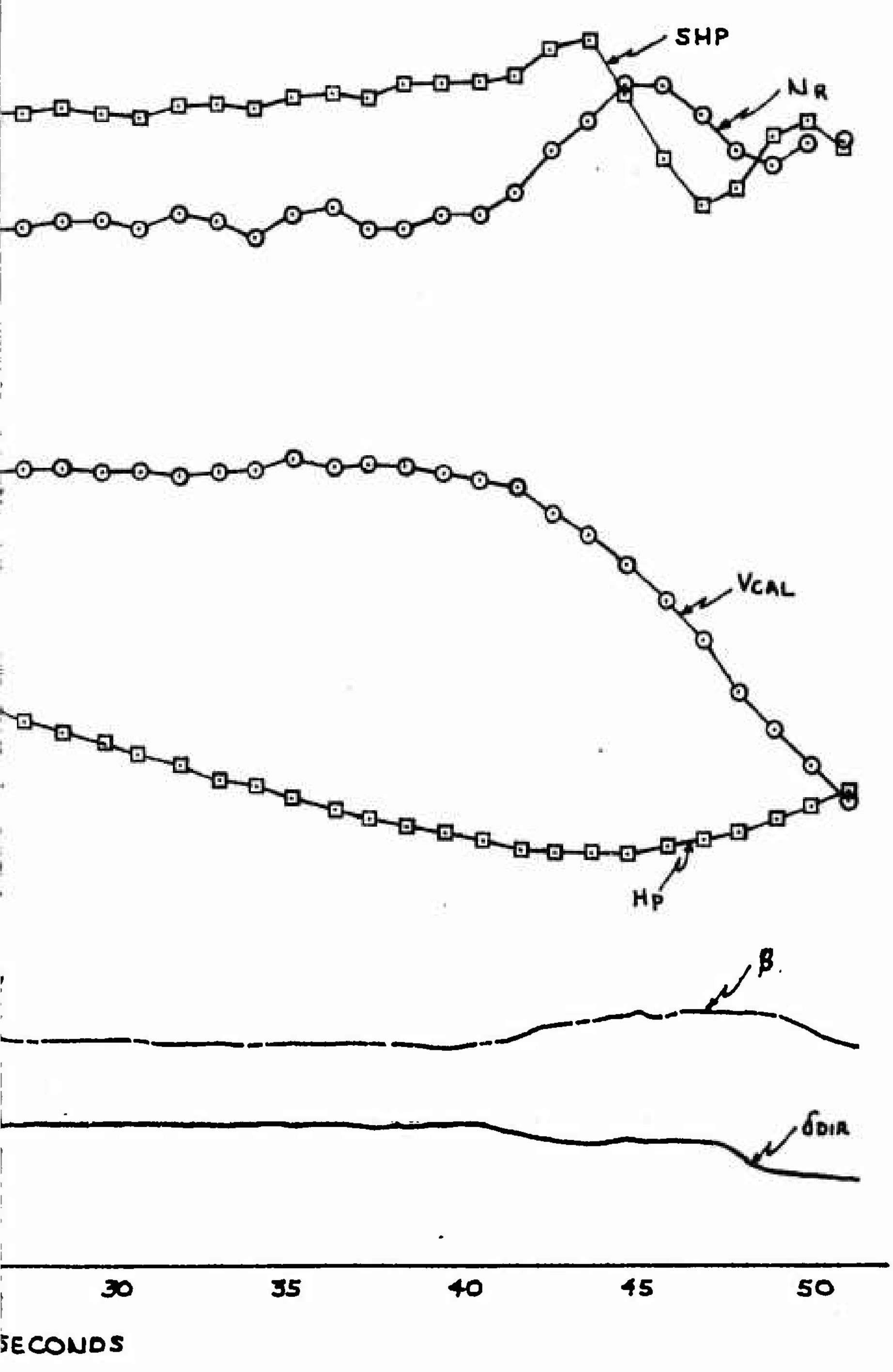


FIGURE 35 (CONTINUED)



B

FIGURE 36

DIVING FLIGHT AND PULL UP MAN

AH-1G USA S/N 615247

HVY. HOG CONFIGURATION WITH ROCKET POD FA

GRWT = 9255 LB. LONG. C.G. POSITION = 195.9 INC

NOTES: 1. COLLECTIVE CONTROL POSITION REMAINED FIXED THROU

2. FAIRED DATA WAS OBTAINED FROM OSCILLOGRAPH

3. SYMBOLIZED DATA WAS OBTAINED FROM PHOTO PANEL

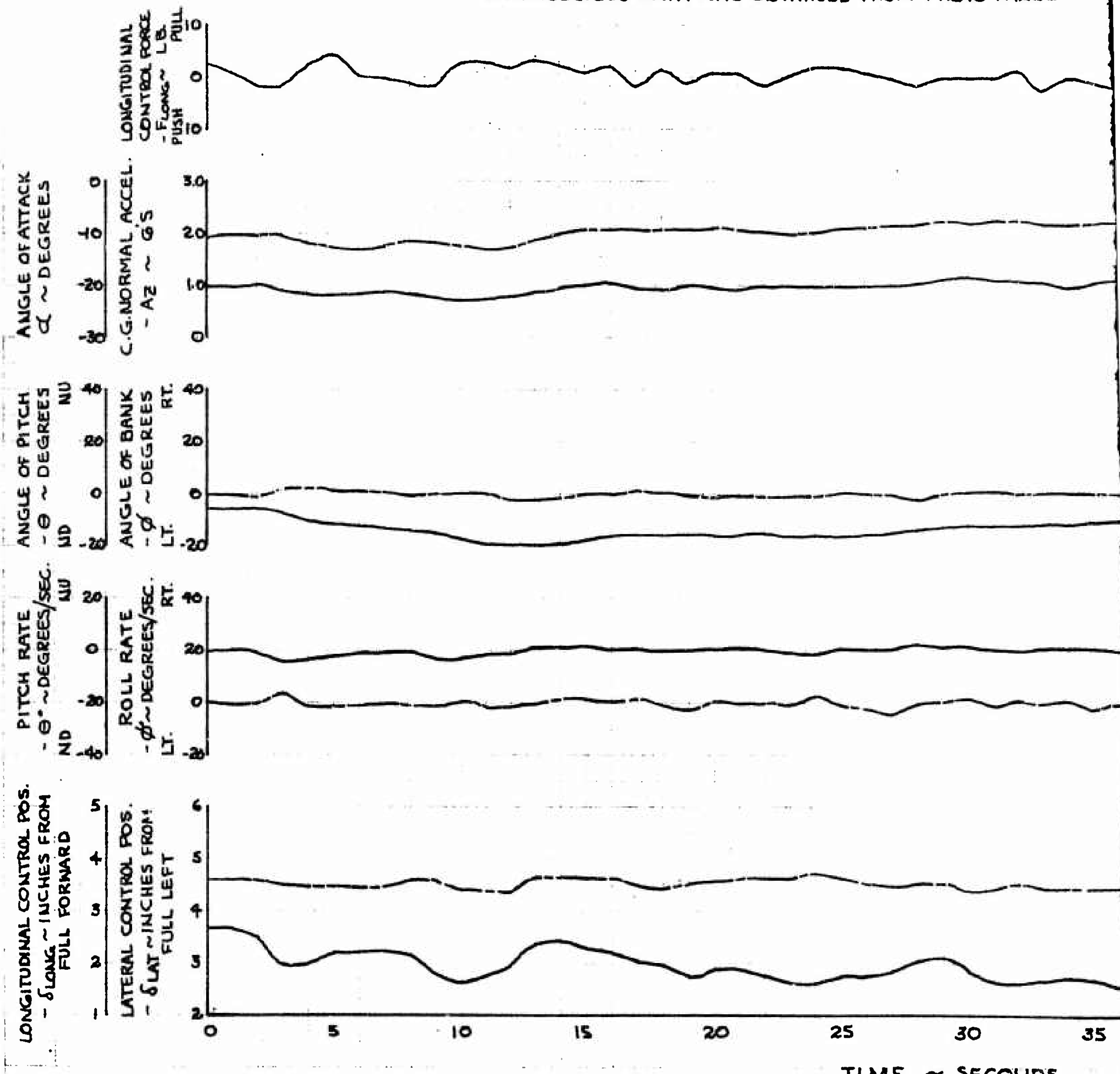


FIGURE 36

AND PULL UP MANEUVER

USA S/N 61 S 2[illegible]7

WITH ROCKET POD FAIRINGS REMOVED

POSITION = 195.9 INCHES (MID)

MAINED FIXED THROUGHOUT ENTIRE MANEUVER

OSCILLOGRAPH

OM PHOTO PANEL

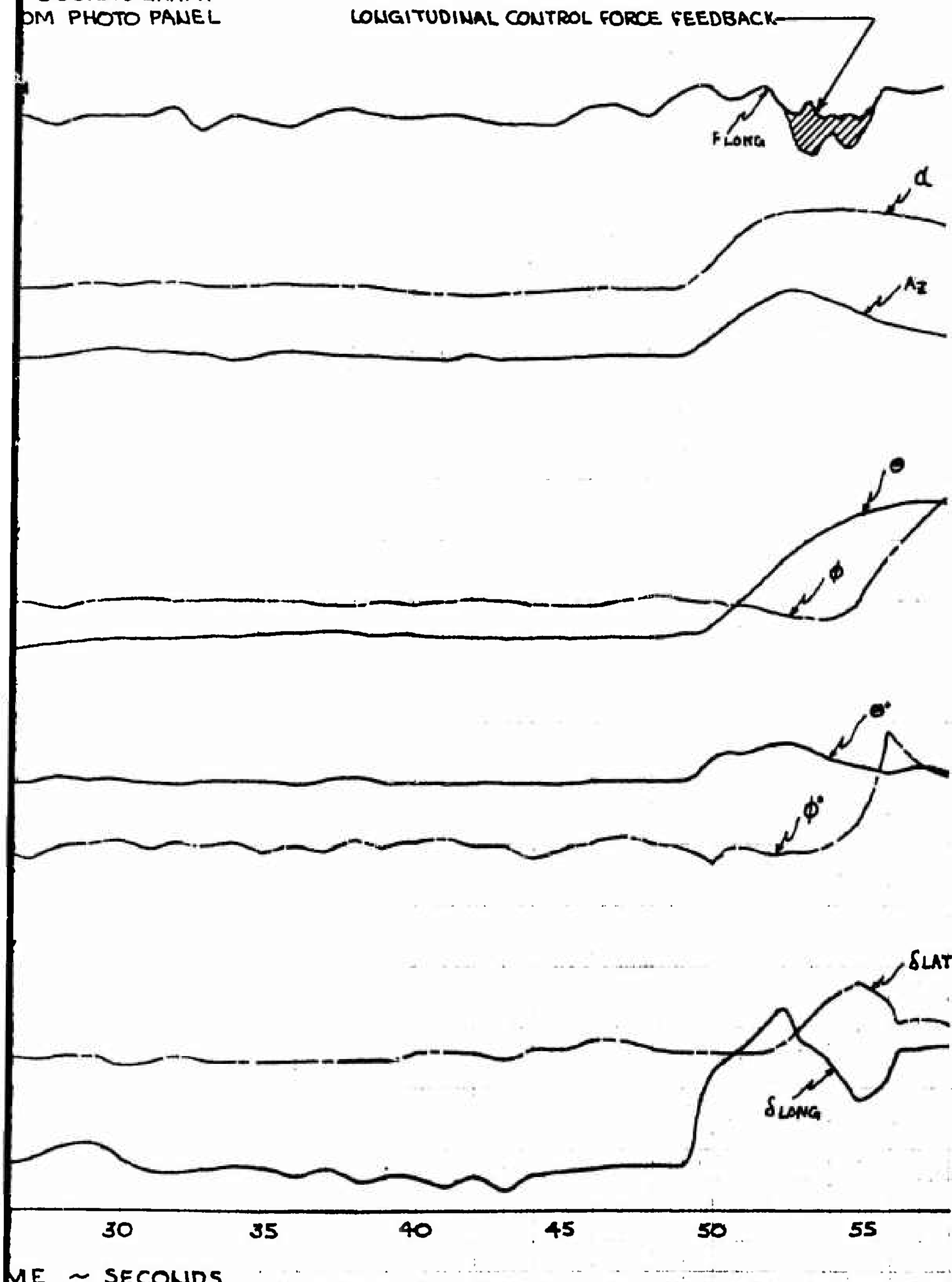
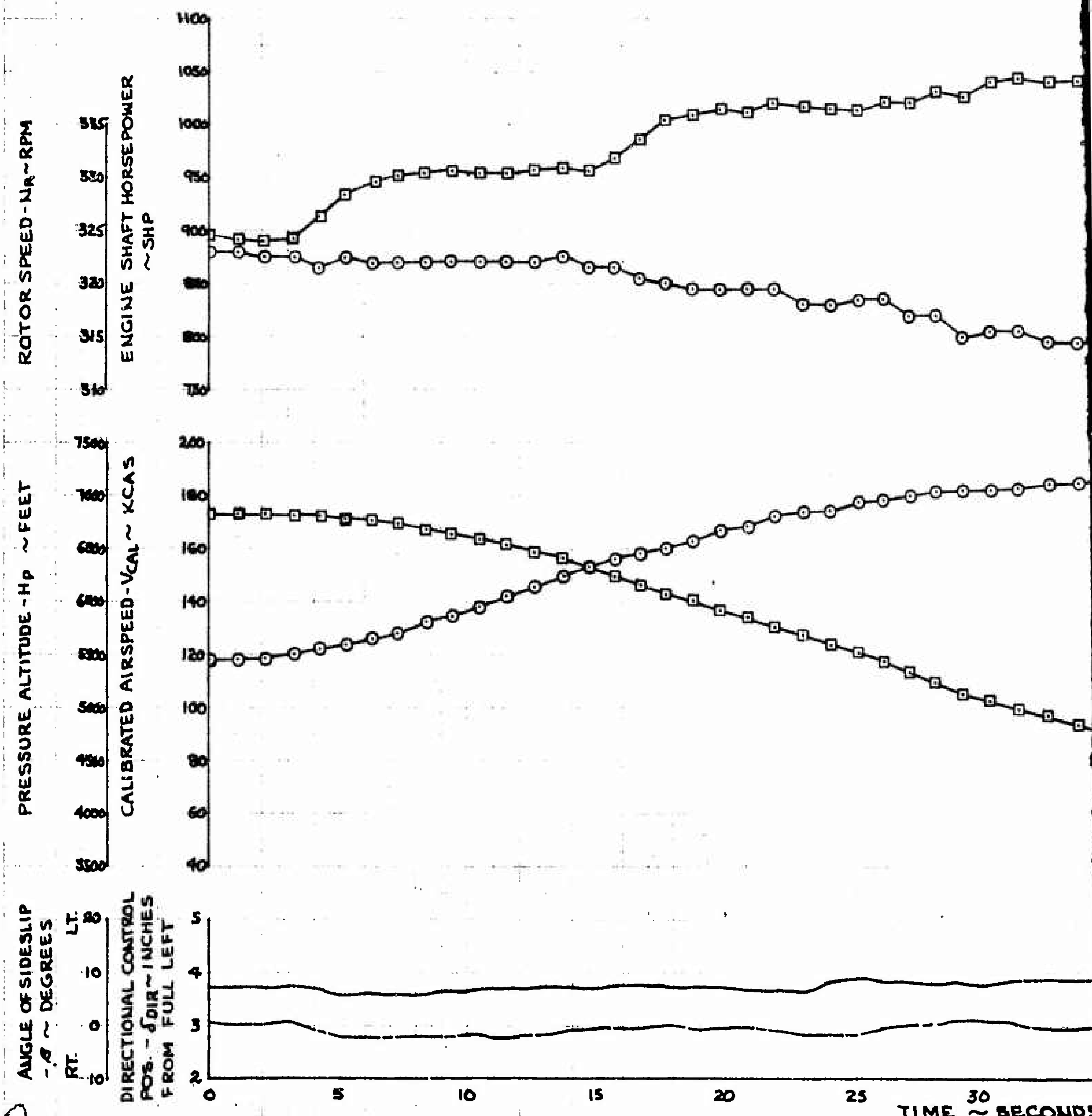


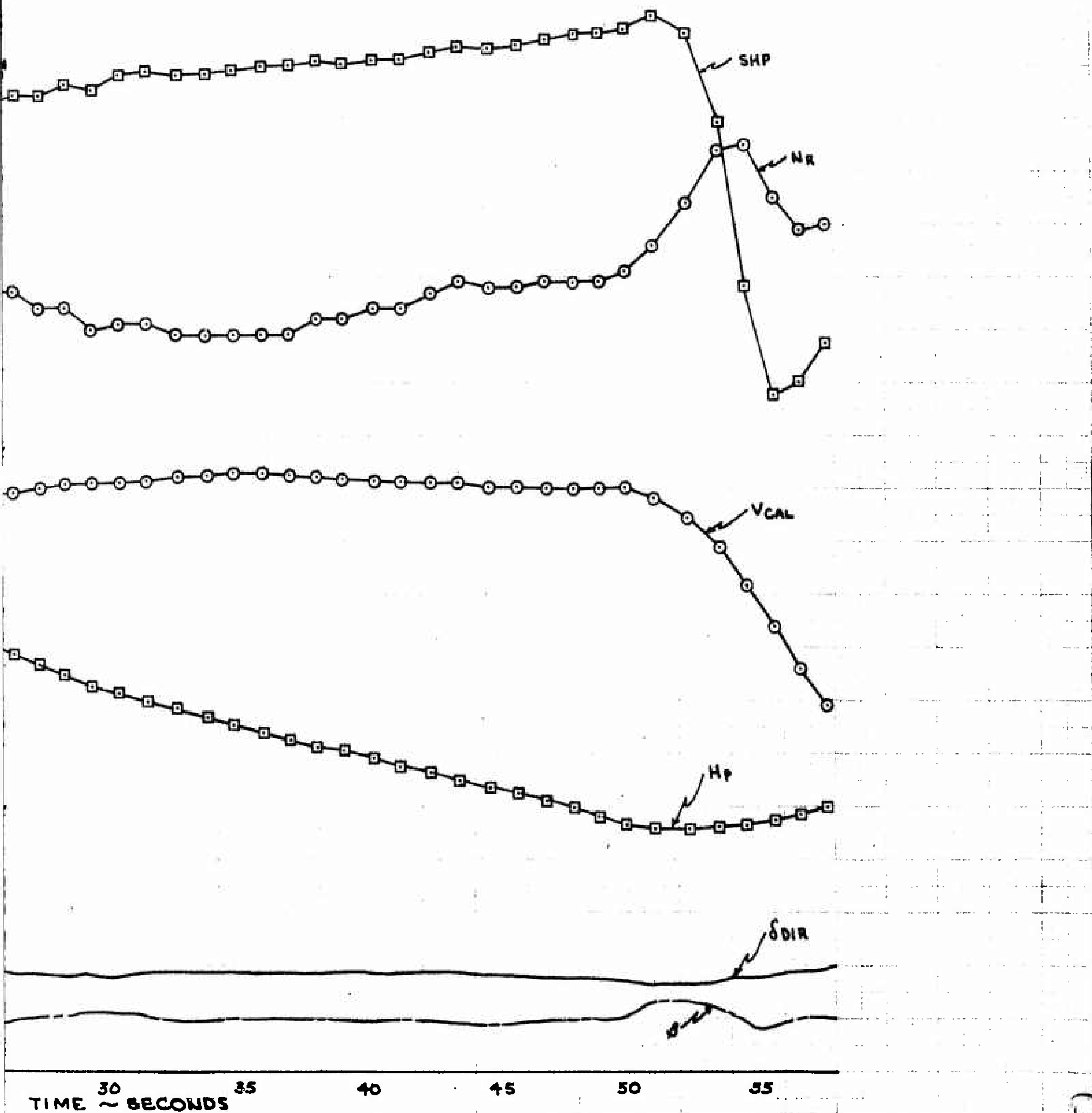
FIGURE 36



ROTOR SPEED - N_R ~ RPM
ENGINE SHAFT HORSEPOWER ~SHP
PRESSURE ALTITUDE - H_P ~FEET
CALIBRATED AIRSPEED - V_{CAL} ~ KCAS
ANGLE OF SIDESLIP - β ~ DEGREES
LT.
RT.
DIRECTIONAL CONTROL POS. - δ_{DIR} ~ INCHES FROM FULL LEFT
TIME ~ SECONDS

A

FIGURE 36 (CONTINUED)



B

APPENDIX VI. SYMBOLS AND ABBREVIATIONS

Abbreviation	Definition	Unit
ALT	Altitude	foot
AVG	Average	--
CG, cg	Center of gravity	--
CONF	Configuration	--
DEG, deg	Degree(s)	degree
DWN	Down	--
EGT	Engine exhaust gas temperature	°C
fig., figs.	Figure, figures	--
FLT	Flight	--
ft	Foot, feet	foot
FS	Fuselage station	inch
fwd	Forward	--
g	Gravitational constant	ft/sec^2
GRWT, grwt	Gross weight	pound
IGE	In ground effect	--
in.	Inch, inches	inch
KCAS	Knots calibrated airspeed	knot
KIAS	Knots indicated airspeed	knot
KTAS	Knots true airspeed	knot
LB, lb	Pound, pounds	pound
LT	Left	--

Abbreviation	Definition	Unit
LONG.	Longitudinal	--
MAX, max	Maximum	--
MIN, min	Minimum	--
NACA	National Advisory Committee Aeronautics	--
ND	Nose down	--
NU	Nose up	--
NO., no.	Number	--
PSI, psi	Pound(s) per square inch	$lb/in.^2$
ref	Reference, referred	--
RPM, rpm	Revolution(s) per minute	rpm
RT	Right	--
SCAS	Stability and control augmentation system	--
SEC, sec	Second	--
SHP, shp	Shaft horsepower	--
S/N	Serial number	--
STD, std	Standard	--
SYM	Symbol	--
WT	Weight	pound

Symbol	Definition	Unit
dE/dt	Rate of energy change	ft-lb/sec
F_{LONG}	Longitudinal cyclic control force	lb

Symbol	Definition	Unit
H_D	Density altitude	foot
HP_{TR}	Tail rotor horsepower	hp
N_R	Main rotor speed	rpm
N_Z	CG normal acceleration	g
R/D	Rate of descent	ft/min
T	Time	sec
V_c	Calibrated airspeed	knot
V_H	Maximum airspeed for level flight	knot
V_L	Limit airspeed	knot
V_T	True airspeed	knot
°C	Degree(s, centigrade	degree
%	Percent	--
α	Angle of attack	degree
β	Angle of sideslip	degree
Δ	Difference	--
γ	Flight path angle	degree
δ_{COLL}	Collective control position	inch
δ_{DIR}	Directional control position	inch
δ_{LAT}	Lateral cyclic control position	inch
$\delta_{Lat_{SCAS}}$	Lateral stability and control augmentation position	percent
θ	Aircraft pitch attitude	degree

Symbol	Definition	Unit
$\dot{\theta}$	Aircraft pitch rate	deg/sec
ϕ	Aircraft roll attitude	degree
$\dot{\phi}$	Aircraft roll rate	deg/sec

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13. ABSTRACT

The Phase D Airworthiness and Qualification tests of the AH-1G helicopter were conducted in California at Edwards Air Force Base and auxiliary test sites during the period 13 June 1968 through 29 July 1969. This addendum to the performance report presents the results of turning performance, in-ground-effect (IGE) level acceleration and deceleration performance and dive recovery tests. These three tests were conducted to validate portions of the AH-1G operator's manual (TM 55-1520-221-10) and enhance the knowledge of interested government agencies as to the limitations and capabilities of the AH-1G helicopter. There were no additional deficiencies or shortcomings revealed by the results of these tests that had not been previously mentioned in Part 1, Part 2 and Part 3 of this report. Three major limitations were encountered during testing that restricted the pilot from achieving maximum performance: 1) level acceleration and deceleration performance IGE is limited by extreme pitch attitudes; 2) level deceleration performance is limited by the pilot's ability to maintain rotor speed below the maximum limit (339 rpm); 3) the cyclic control feedback limits aircraft turning and dive recovery performance at heavy gross weight and/or when high load factors are encountered.

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14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
AH-1G helicopter Addendum Performance report Turning performance In-ground-effect level acceleration Deceleration performance Dive recovery						

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